



# Flight Test of Propulsion Monitoring and Diagnostic System

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# Table of Contents

Executive Summary .....	1
Section 1. Introduction.....	2
1.1 Background .....	2
1.2 AGATE Program Results .....	2
1.3 Program Overview .....	3
1.4 PMDS Overview .....	3
1.5 Relevance to Aviation Safety.....	4
1.6 PMDS Background Technical Data.....	5
Section 2. Flight Testing .....	6
2.1 Flight Test Aircraft .....	6
2.2 PMDS Installation.....	7
Section 3. Data Analysis .....	9
3.1 Background .....	9
3.2 Flight Test Overview .....	9
3.3 Data Collection .....	10
3.4 Static Correlation Modeling.....	11
3.5 Linear Dynamic Modeling.....	17
3.6 Nonlinear Dynamic Modeling .....	30
3.7 Data Analysis Conclusions .....	40
Section 4. Technology Roadmap .....	42
4.1 Modeling and Fault Diagnosis Algorithms Development .....	42
4.2 Hardware and Software Development .....	48
4.3 System Development .....	53
Section 5. New Technology .....	56
Section 6. Conclusions and Recommendations .....	57
Appendix .....	

## List of Figures

Figure 1-1. PMDS .....	4
Figure 2-1. Flight Test Aircraft.....	6
Figure 2-2. PMDS Flight Test Arrangement .....	7
Figure 2-3. PMDS Hardware .....	7
Figure 2-4. PMDS Installation.....	8
Figure 3-1. PMDS Hardware Arrangement .....	10
Figure 3-2. EGT Fault Model Results, Flight 44 .....	13
Figure 3-3. CHT2 Measured Data and Modeling Data.....	14
Figure 3-4. CHT Fault Model Results, Flight 41 .....	15
Figure 3-5. CHT Fault Model Results, Flight 44.....	16
Figure 3-6. Engine Bay Temperature Sensor Fault Model Results, Flight 44.....	17
Figure 3-7. Linear Dynamic Model Data.....	20
Figure 3-8. Linear Dynamic Model Results, Flight 41 .....	22
Figure 3-9. Linear Dynamic Model Results, Flight 44 .....	26
Figure 3-10. Nonlinear Dynamic Model Data .....	31
Figure 3-11. Nonlinear Dynamic Model Results, Flight 41 .....	33
Figure 3-12. Nonlinear Dynamic Model Results, Flight 44 .....	35
Figure 3-13. Nonlinear vs. Linear Model Results, Flight 41 .....	37
Figure 3-14. Nonlinear vs. Linear Model Results, Flight 44 .....	39
Figure 4-1. Modeling and Fault Detection.....	42
Figure 4-2. Oil Pressure Failure Detection and Warning Diagnosis .....	44
Figure 4-3. Modeling and Fault Diagnosis Algorithms Development Roadmap .....	47
Figure 4-4. PMDS Architecture .....	48
Figure 4-5. PMDS Hardware Architecture .....	49
Figure 4-6. PMDS Hardware .....	50
Figure 4-7. Hardware and Software Development Roadmap.....	52
Figure 4-8. System Development Roadmap .....	55

## List of Tables

Table 1-1. PMDS Glossary .....	5
Table 3-1. Flight Tests .....	9
Table 3-2. Linear Model Data.....	21
Table 3-3. Linear Model Plots .....	21
Table 3-4. Nonlinear Model Data .....	32
Table 3-5. Nonlinear Model Plots.....	32
Table 3-6. Comparison Plots.....	36
Table 4-1. Oil Pressure Fault Modes and Corresponding Detection Tests.....	45

## Executive Summary

This report presents the results of the NASA program entitled “Flight Test of Propulsion Monitoring and Diagnostic System,” performed by Honeywell and Aurora Flight Sciences. The objective of this program was to build on the results of the propulsion monitoring and diagnostic system (PMDS) technology developed by Honeywell under the NASA Advanced General Aviation Transport Experiment (AGATE) program and apply them to the broader goals of the NASA Aviation Safety program. The technical work included two flight tests using PMDS equipment and analysis of flight test data. The target application for the PMDS is piston-engine-driven general aviation aircraft.

The PMDS concept is intended to independently monitor the performance of the engine, providing continuous status to the pilot along with warnings if necessary. Specific sections of this data would be available to ground maintenance personnel via a special interface. The inputs to the PMDS include the digital engine controller sensors and other sensors. At its present stage of development, the PMDS monitors and records engine parameters and stores them into an engine history database for subsequent processing by off-line diagnostic algorithms. At present, the system does not compare the parameter values with engine norms to perform on-line diagnostics and prognostics (this extended functionality would be added in future development).

Technological advances in sensing, processing, and software have resulted in more affordable and more capable health monitoring technology. The application of health monitoring technology to aircraft engines has tremendous potential given the complexity, harsh environmental conditions, and natural degradation that this machinery exhibits. Benefits include increased safety and reliability and reduced operating costs.

The technical work performed on this program provided the following key results:

- It demonstrated the ability of the PMDS to detect a class of selected sensor hardware failures.
- It demonstrated the ability of the PMDS hardware to successfully model the engine for the purpose of engine diagnosis. Not surprisingly, nonlinear dynamic models performed better than linear dynamic models for the same number of inputs and states.

Future development work for an engine monitoring and diagnostic system should employ the following elements:

- Engine/aircraft modeling should combine first-principles and empirical approaches. Empirical methods can be used to calibrate unknown parameter values as needed.
- The monitoring and diagnostic system should employ additional inputs outside the engine, such as aircraft speed, aileron, elevator, rudder, and flap settings, propeller pitch, etc.
- A prioritized list of engine faults is needed to guide the diagnostic development work.

The monitoring and diagnostic system should be able to gather input data from the full authority digital engine controller (FADEC) and other systems in the aircraft over a digital avionics bus. This data sharing capability will enable the use of more sophisticated models and will help to minimize the installed cost of the PMDS.

## **Section 1. Introduction**

This report presents the results of the NASA program entitled “Flight Test of Propulsion Monitoring and Diagnostic System,” performed by Honeywell and Aurora Flight Sciences. The following sections present the detailed technical results along with a set of conclusions and recommendations based on experience gained in performing the work.

### **1.1 Background**

A propulsion monitoring and diagnostic system (PMDS) can provide valuable benefits to general aviation users. The PMDS provides increased confidence in the propulsion system while in flight for improved safety and provides valuable diagnostic data to ground maintenance technicians for reduced maintenance costs. The target application for the PMDS is piston-engine-driven general aviation aircraft. In this program, the PMDS technology developed by Honeywell under the NASA Advanced General Aviation Transport Experiment (AGATE) program was extended through additional flight testing and data analysis to demonstrate its current capabilities.

The purpose of the PMDS concept is to provide the pilot with engine health indications and to inform the pilot when the engine requires preventative maintenance. By providing this information before in-flight failure of the engine, it greatly enhances flight safety and provides simplified engine diagnostics for the pilot. The technology developed under AGATE funding is the core of a future fully functional PMDS. The hardware and software developed under AGATE funding monitors and records engine parameters and stores them in an engine history database for subsequent processing by off-line diagnostic algorithms. At present, the unit does not compare the parameter values with engine norms to perform on-line diagnostics and prognostics (this extended functionality would be added in future development).

### **1.2 AGATE Program Results**

Honeywell was a participant in the AGATE Propulsion Sensors and Controls Work Package. During 1999, Honeywell built the PMDS hardware, implemented the firmware that records and preprocesses the flight data, and with the assistance of Aurora Flight Sciences Corporation, performed an initial flight test. A simple engine model, developed to postprocess the flight test data, verified that we could detect a failed (disconnected) exhaust gas temperature (EGT) sensor signal. A shortfall in funding prevented further test flights under the AGATE program. However, these results showed the significance of the core PMDS in that it provides both hardware and software design guidelines for successfully interfacing the PMDS to general aviation aircraft.

Additional test flights and data analysis were performed on this 2001 NASA program to verify that the design and integration of the core PMDS into the aircraft is a suitable base on which to build the diagnostic capability.



### 1.3 Program Overview

The objective of this program was to build on the results of the AGATE work and apply them to the broader goals of the NASA Aviation Safety program. The technical work included two flight tests using PMDS equipment and analysis of flight test data.

Our technical approach for the flight testing and data analysis consisted of four steps:

1. Collect data from a set of pertinent engine sensors during a baseline test flight (no failures).
2. Compute an engine model based on a least-squares fit to flight test data.
3. During the second test flight, introduce sensor faults to test the diagnostic algorithm.
4. Postprocess the flight data to diagnose the faulty sensor/variable.

This program also included the development of a roadmap detailing the recommended next steps in applying the PMDS technology to general aviation.

### 1.4 PMDS Overview

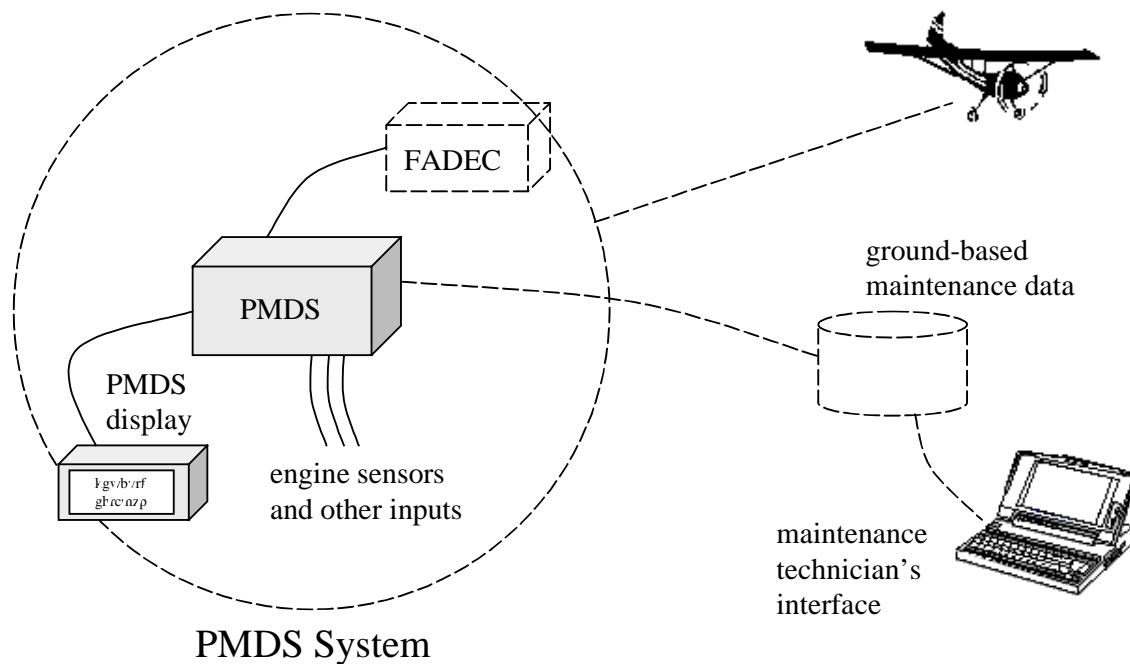
The PMDS is a separate subsystem designed to independently monitor the performance of the engine, providing continuous status to the pilot along with warnings if necessary. Specific sections of this data are available to ground maintenance personnel via a special interface. The PMDS also provides a set of data for maintenance event prediction to be used by ground personnel or for possible impending failure information to be displayed to the pilot. The PMDS will continuously monitor its own performance to ensure its own integrity and capability.

The PMDS will be able to detect and diagnose the most common engine failures. The set of failures to be detected will be defined in future development phases. A top level of the system is shown in Figure 1-1.

The PMDS continuously monitors the performance of the engine in order to detect failures and predict impending failures (prognosis). Failures and warnings of impending failures are indicated to the pilot, and the collected engine diagnostic information pertaining to the failures and/or warnings is also available to a ground maintenance technician. The AGATE PMDS development effort defined the following performance goals for the system:

- Early detection time: The PMDS is intended to detect and indicate a warning of impending failure at least 8 flight hours prior to failure.
- High probability of detection and coverage: The PMDS is intended to detect 90% of impending failures in the engine.

The inputs to the PMDS include the full authority digital engine controller (FADEC) sensors and other engine sensors. The outputs consist of the pilot warning display and the maintenance device interface. The maintenance device interface could be a hand-held interrogation and display device that would allow maintenance personnel to determine the status of the engine (as



**Figure 1-1. PMDS**

well as the PMDS itself) and define any necessary maintenance actions. Readout capability is included in the on-board maintenance panel; however, some installations could forgo this panel, depending completely on ground-based equipment that would receive data from the interface port.

Sensor inputs will be received via digital interfaces from the FADEC, single-lever power control (SLPC), and other digital systems on the aircraft, or via other sensors directly connected to the PMDS. Examples of these sensor inputs are as follows:

- FADEC: exhaust gas temperature (EGT), cylinder head temperature (CHT), engine speed (RPM), oil pressure, etc.
- SLPC: engine power lever position, throttle command, propeller pitch command, etc.
- Other inputs via an avionics bus: air speed, pilot inputs to control surfaces, etc.
- Other sensors directly connected to the PMDS: vibration sensors, oil particle sensors, etc.

A glossary of terms relating to the PMDS is shown in Table 1-1.

## **1.5 Relevance to Aviation Safety**

Technological advances in sensing, processing, and software have resulted in more affordable and more capable health monitoring technology. The application of health monitoring technology to aircraft engines has tremendous potential given the complexity, harsh

**Table 1-1. PMDS Glossary**

Early Detection:	System predicts that failure will occur in near future, lights warning light.
Failure Detection:	System detects that failure has occurred, lights failure light, (future implementations may take backup action if indicated).
False Alarm	System indicates that failure has occurred, although in fact, indicated failure has not occurred.
Impending Failure	Engine condition has changed so as to cause a failure in near future.
Diagnostics	The process by which a particular fault mode is indicated.
Detection Time	The time between the occurrence of either an impending failure (in the case of early detection) or a failure (in the case of failure detection) and the corresponding failure or warning indication.
Accuracy	RSS (Root Sum Square) value including scale factor tolerance, linearity, offset and temperature effects.
Failures	A failure is a fault which will require ground maintenance action to correct.
FADEC	Full Authority Digital Engine Controller
SLPC	Single Lever Power Control
EGT	Exhaust Gas Temperature
CHT	Cylinder Head Temperature

environmental conditions, and natural degradation that this machinery exhibits. Benefits include increased safety and reliability and reduced operating costs.

The benefits of engine monitoring and diagnostics will be valuable to all segments of general aviation, from the individual aircraft owner to large fleet operators. All of these potential users share a common interest in having a capability to increase the availability of their aircraft engines. At present, we don't know of any commercially available engine monitoring systems for piston engine aircraft. However, these types of devices are currently offered as optional equipment in at least one single-engine turboprop aircraft (the Pilatus PC-12). Pilatus offers an "engine trend monitoring" option with the PC-12 as described at [http://www.pilatus-aircraft.com/2\\_ga\\_commercial/frameset\\_pc12.htm](http://www.pilatus-aircraft.com/2_ga_commercial/frameset_pc12.htm).

## **1.6 PMDS Background Technical Data**

Under the AGATE program, Honeywell prepared a set of technical documents for the PMDS concept. These documents define the system requirements, modeling methods, and fault detection and failure diagnosis methods. This body of information serves as a baseline for future development of the system.

The PMDS concept is intended to meet standard commercial avionics integration requirements that apply to the intended application configurations. The PMDS concept is also intended to conform to the same environmental, electrical, and mechanical standards that apply to comparable commercial avionics equipment.

## Section 2. Flight Testing

The flight testing was performed by Aurora Flight Sciences as described in the following subsections. Additional information is presented in the Aurora flight test report in the Appendix.

### 2.1 Flight Test Aircraft

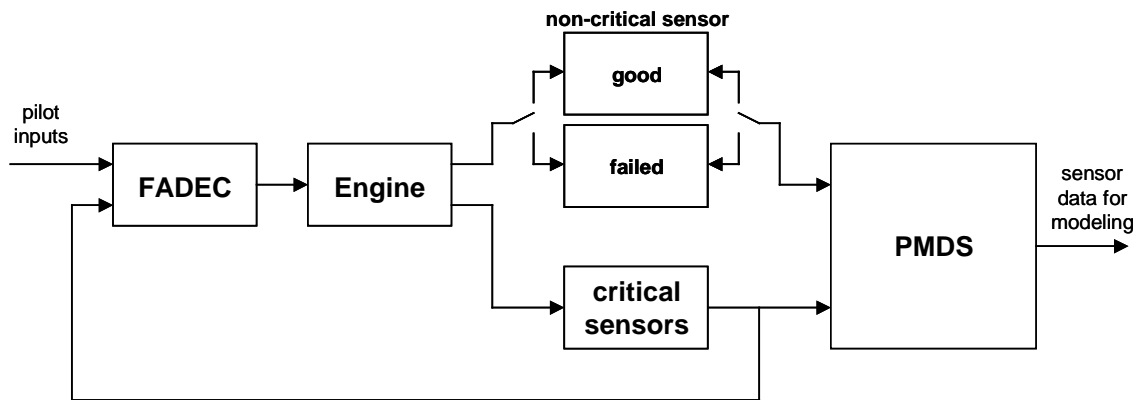
The flight tests were performed using Aurora's twin engine Cessna O-2 Chiron aircraft, as shown in Figure 2-1. This aircraft was equipped with a SLPC and FADEC controlling the front engine (the rear engine was not involved in the flight testing). With the SLPC and FADEC, electric servo actuators control the throttle and the prop governor; the pilot commands a single thrust command. Electronic port fuel injection, electronic ignition system with several redundant feedback loops controls mixture and thermal control (CHT, EGT, exhaust gas oxygen).



**Figure 2-1. Flight Test Aircraft**

The Chiron front engine (monitored during this test) is a Teledyne IO-360ES. The flight tests were performed according to flight test plans developed jointly by Honeywell, Aurora, and NASA. Flight test data was logged on the PMDS. After each flight, the PMDS data was downloaded via serial interface to a PC and sent to Honeywell via the Internet for data reduction and analysis.

To test the PMDS diagnostic concept, we simulated a sensor failure by temporarily disconnecting some noncritical sensors on one of the flights. The flight test approach is shown in Figure 2-2.



**Figure 2-2. PMDS Flight Test Arrangement**

In addition, we noticed some unplanned intermittent sensor faults as described in Section 3 of this report. This flight testing procedure had no effect on the engine's performance, since the FADEC recognizes and accounts for faulty EGT signals.

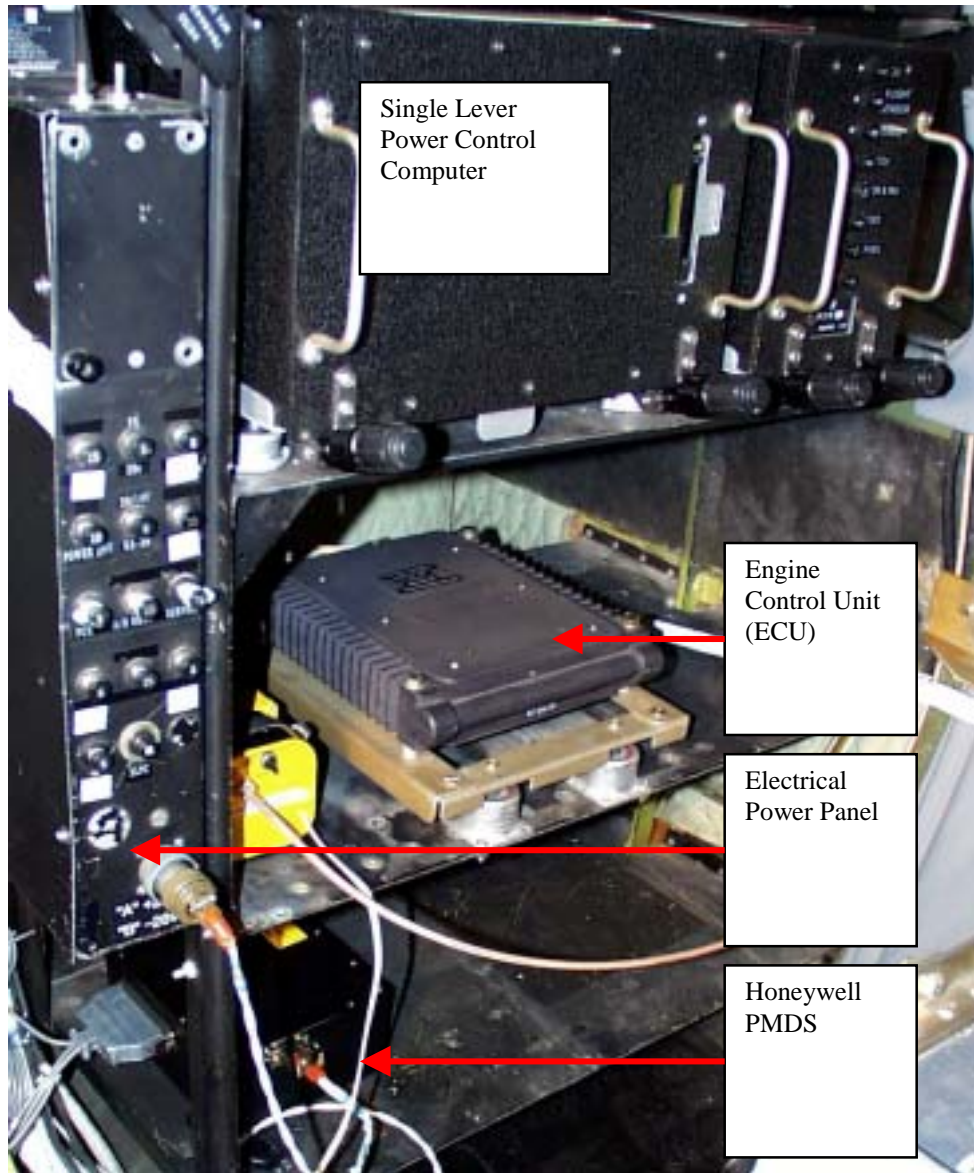
## 2.2 PMDS Installation

The Honeywell PMDS was mounted in the cabin of the aircraft. Figures 2-3 and 2-4 show the PMDS hardware and its installation in the aircraft.



**Figure 2-3. PMDS Hardware**





**Figure 2-4. PMDS Installation**

## Section 3. Data Analysis

The flight test data files were received from Aurora Flight Sciences and prepared for analysis. Several technical analyses were performed on the data using Matlab scripts. This technical analysis work is described in the following subsections.

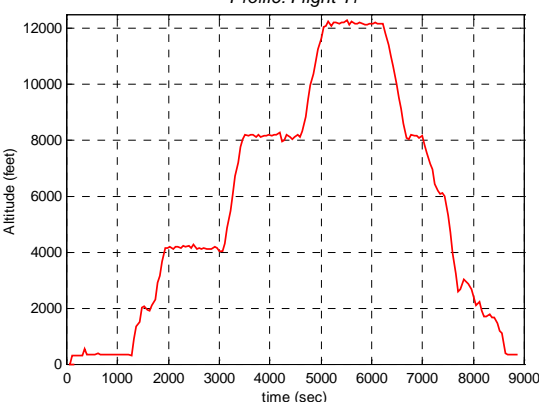
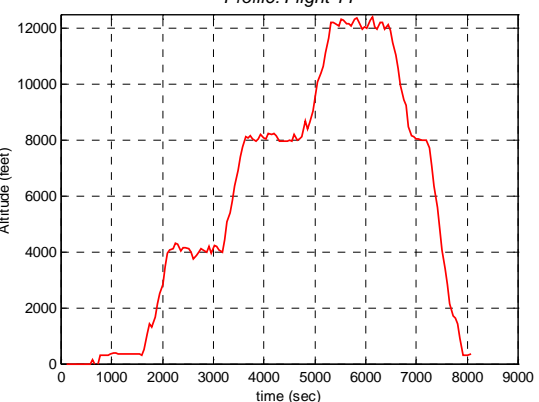
### 3.1 Background

The flight testing was performed by Aurora Flight Sciences in a manner similar to that used for the earlier AGATE work. Aurora's twin engine Cessna O-2 Chiron aircraft was used for these tests. The Honeywell PMDS was employed with Aurora's FADEC and electronic SLPC, which controlled the front engine in the aircraft. The PMDS collected various engine data and other parameters made available directly from the FADEC via a digital communications link.

### 3.2 Flight Test Overview

Under this program, two flight tests of the PMDS were conducted. Data was collected at a variety of altitudes and power settings over the operating envelope of the engine. A top-level description of the two flight tests is presented in Table 3-1 below.

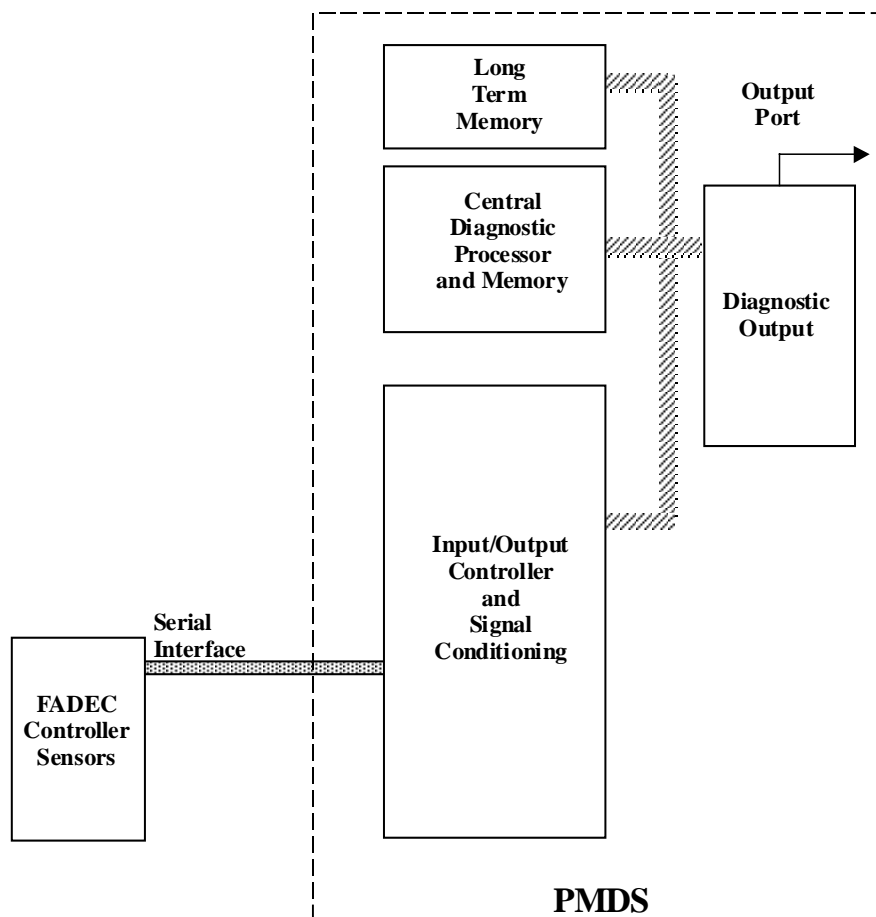
**Table 3-1. Flight Tests**

 <p><i>Profile: Flight 41</i></p>	 <p><i>Profile: Flight 44</i></p>
<p align="center"><u><b>Flight 41</b></u></p> <p>Date: April 25, 2001  Setup: Baseline flight  Simulated Faults: none  Intermittent Faults: CHT2 and EGT4</p>	<p align="center"><u><b>Flight 44</b></u></p> <p>Date: April 27, 2001  Setup: Add simulated faults  Simulated Faults:</p> <ul style="list-style-type: none"> <li>• Engine bay temperature 2</li> <li>• Mass airflow sensor</li> </ul> <p>Intermittent Faults: CHT2, EGT1, and EGT4</p>

The simulated fault in the mass airflow sensor was not used in the analysis because the sensor data was not set up for communication to the PMDS. This was an oversight in the preparation of the test plan, but it did not affect our ability to accomplish the objectives of the flight test and analysis work. The flight test plans and flight test engineer reports are presented in the Appendix.

### 3.3 Data Collection

For this project, the PMDS collected engine performance data from the engine FADEC via a digital communications link. The hardware arrangement is shown in Figure 3-1. (No directly connected engine sensors were used in this project.)



**Figure 3-1. PMDS Hardware Arrangement**

The current PMDS hardware/firmware records the values of 25 variables: time, pressure altitude, air pressure, fuel pressure, oil pressure, fuel temperature, oil temperature, bay 1 temperature, bay 2 temperature, six EGTs, six CHTs, outside air temperature, manifold air pressure, air charge temperature, and engine RPM. The 25 variables are measured at 5 Hz. To average out noise and to fit all the data from many hours of flight into limited memory, the data is reduced in the following way:



1. Over each 51-second time window, a least-squares-error linear fit is made to each variable's time history in that window.
2. The slope and intercept of this linear fit is saved, together with the maximum and minimum values of each variable in this window.

A list of the sensor data collected from the FADEC is presented in the Appendix.

### 3.4 Static Correlation Modeling

A static correlation model can be used to estimate the value of one or more sensors using measured data from other sensors. While this is not a dynamic model per se, it does employ measured data across a wide range of operating conditions and is capable of producing a good estimate of the sensor values of interest. These estimated values can then be compared to measured sensor data for monitoring and diagnostic purposes. In practice, it would be beneficial to create several static correlation models using different combinations of inputs in case any one of the inputs is faulty (thereby causing all of the outputs to be invalid). This approach would enable the system to check each model to determine which input is faulty.

#### 3.4.1 Static Correlation Modeling Approach

The form of the static correlation model is

$$y = f(x)$$

where

$$y \in R^p = \text{output (estimates of desired sensors)}$$

$$x \in R^n = \text{input (measured data for other sensors, including their derivatives)}$$

The model (matrix  $A$ ) is computed using a least-squares approximation to measured data. For estimating the five EGT states, matrix  $A$  is computed from

$$AX = Y$$

where

$$X = \begin{bmatrix} x_1(1) & \cdots & x_1(k) \\ \vdots & & \vdots \\ x_n(1) & \cdots & x_n(k) \end{bmatrix} \in R^{n \times k}$$

$$Y = \begin{bmatrix} EGT_1(1) & \cdots & EGT_1(k) \\ \vdots & & \vdots \\ EGT_p(1) & \cdots & EGT_p(k) \end{bmatrix} \in R^{p \times k} \quad \text{(In this example, } p = 5 \text{ because one of the EGTs was bad in both test flights.)}$$

$k$  = number of samples (period of time) over which the estimate is desired

The correlation matrix  $A$  is computed from measured data as follows:

$$A = YX^T (XX^T)^{-1}$$

### **3.4.2 Static Correlation Modeling of EGT Sensors**

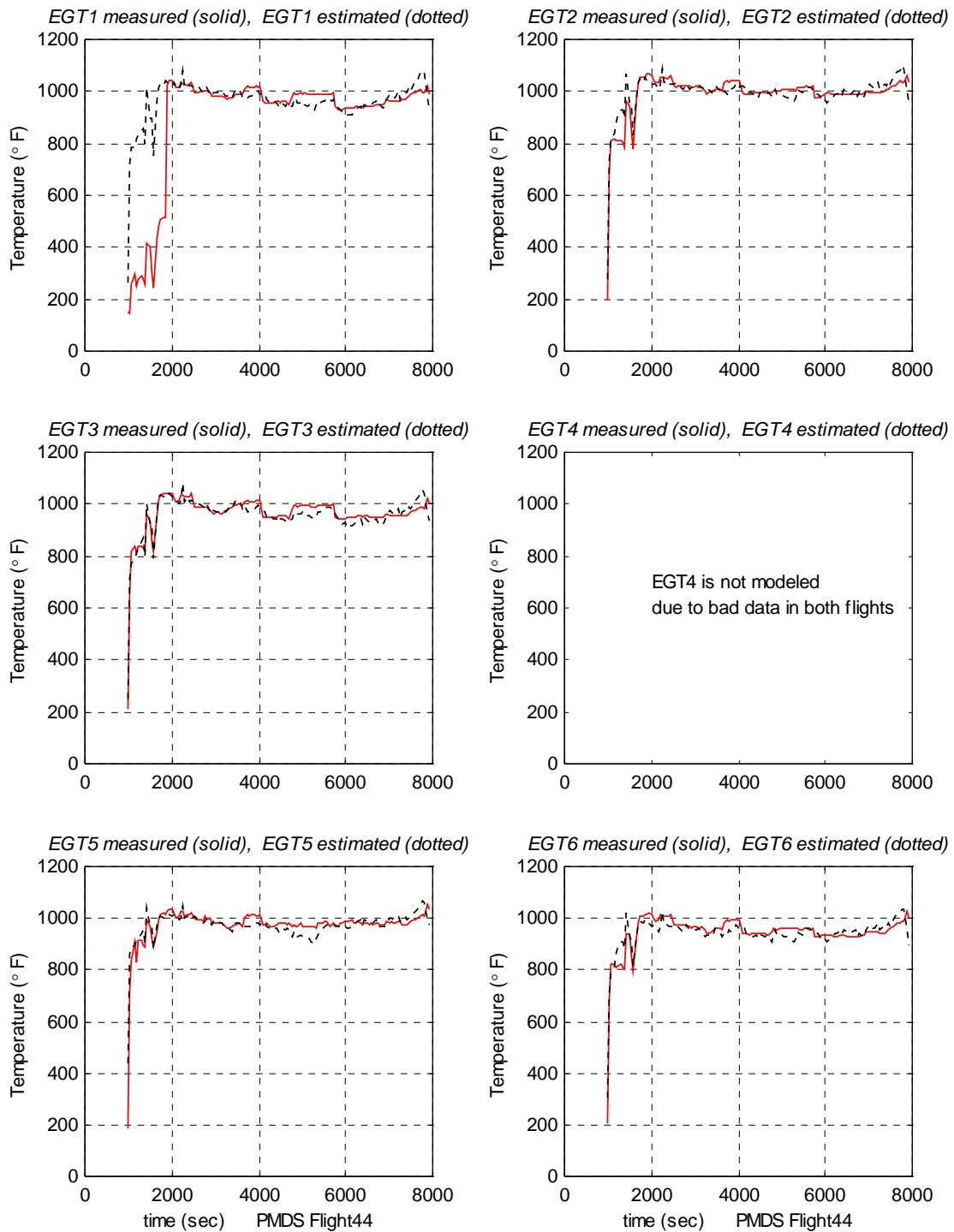
A static correlation model for the EGT sensors was computed using the data from Flight 41 as described above. This model was then applied to measured data from Flight 44 (using sensors other than the EGTs as input data). The resulting estimated EGT data compared very well against the actual measured EGT data for Flight 44, as shown in Figure 3-2. As mentioned earlier, EGT1 in Flight 44 showed some intermittent bad data (i.e., an intermittent fault). This bad data is clearly visible when compared to the estimated values shown in Figure 3-2.

### **3.4.3 Static Correlation Modeling of CHT Sensors**

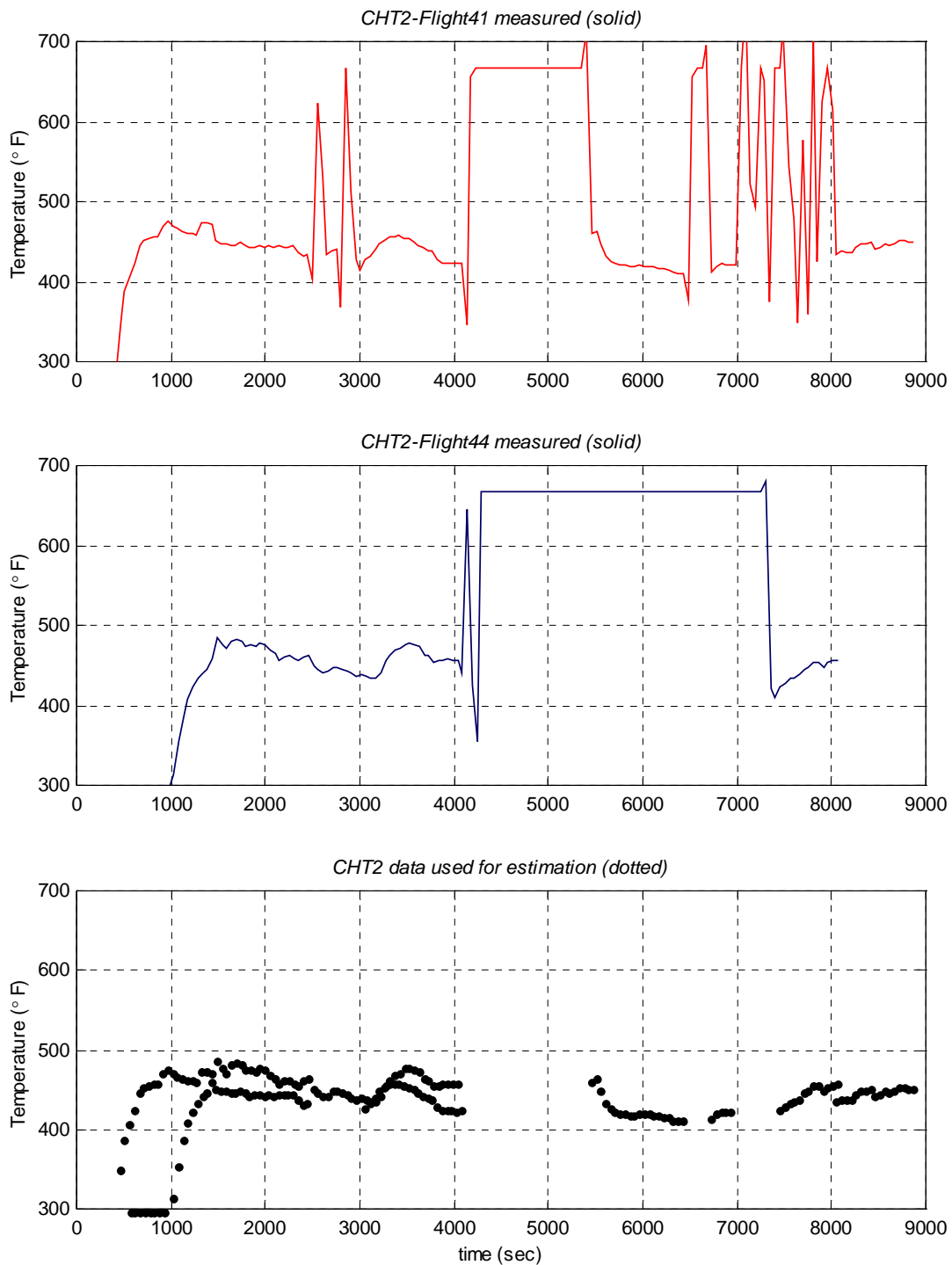
As mentioned earlier, sensor CHT2 showed some intermittent bad data in both Flight 41 and Flight 44. The periods of bad data are shown in the top and middle subplots of Figure 3-3. For modeling purposes, the good data from each flight was combined to create the model. This model input data is shown in the lower subplot of Figure 3-3. A static correlation model for the CHT sensors was computed using this data. This model was then applied to measured data from Flight 41 (using sensors other than the CHTs as input data). The resulting estimated CHT data compared very well against the actual measured valid CHT data for Flight 41, as shown in Figure 3-4. The intermittent fault in CHT2 is clearly visible when compared to the estimated values shown in Figure 3-4. The same analysis was done for Flight 44, which produced similar results as shown in Figure 3-5.

### **3.4.4 Static Correlation Modeling of Engine Bay Temperature Sensors**

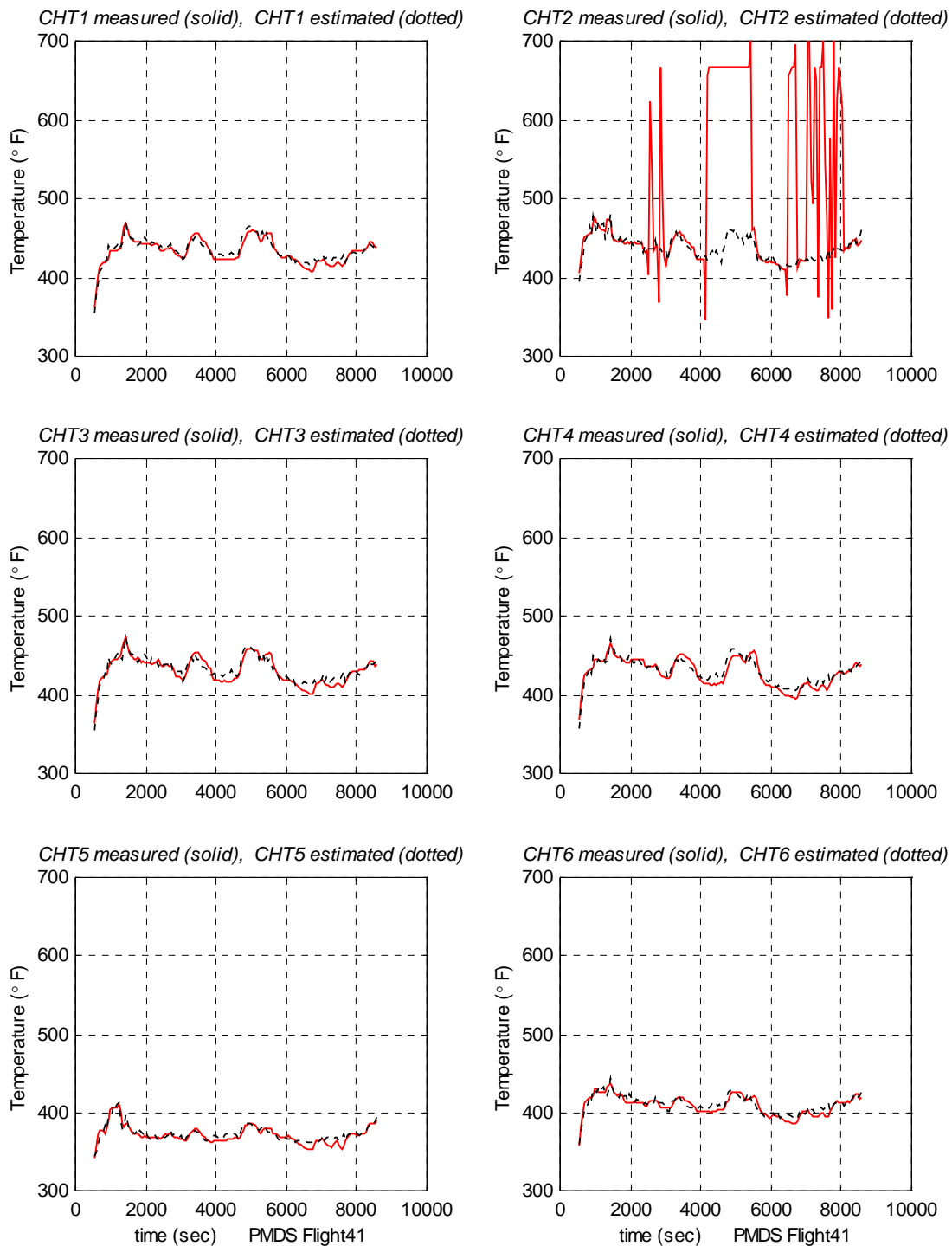
Both engine bay temperature sensors provided valid data in Flight 41. For Flight 44, bay temperature sensor 2 was disconnected to simulate a fault condition. A static correlation model for the bay temperature sensors was computed in the manner described earlier (using measured data from Flight 41). This model was then applied to measured data from Flight 44 (using sensors other than bay temperature sensors as input data). The resulting estimated bay temperature sensor 1 data compared very well against the actual measured data for Flight 44, as shown in the top subplot of Figure 3-6. The simulated fault condition in bay temperature sensor 2 (consistently low) is clearly visible when compared to the estimated values shown in the lower subplot of Figure 3-6.



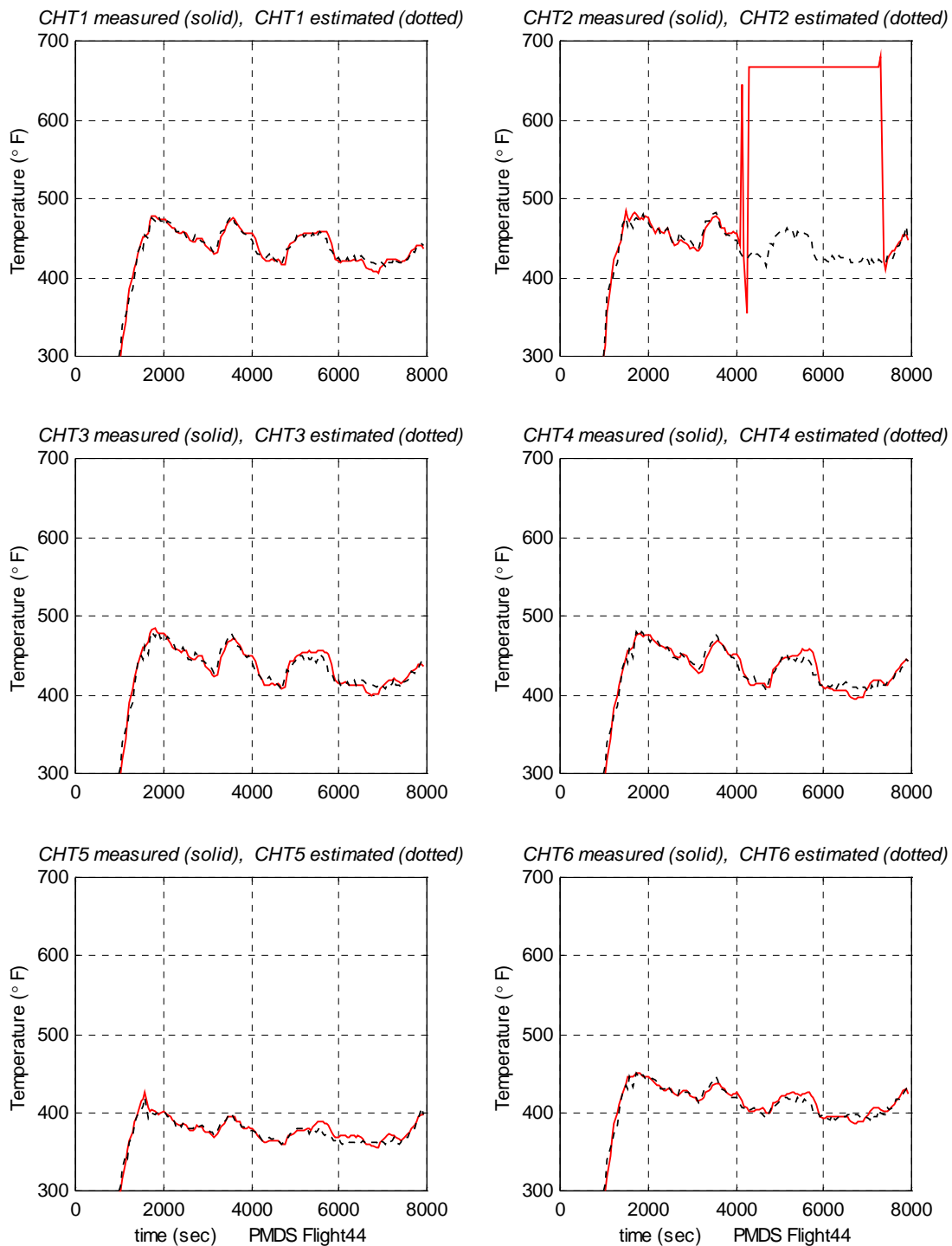
**Figure 3-2. EGT Fault Model Results, Flight 44**



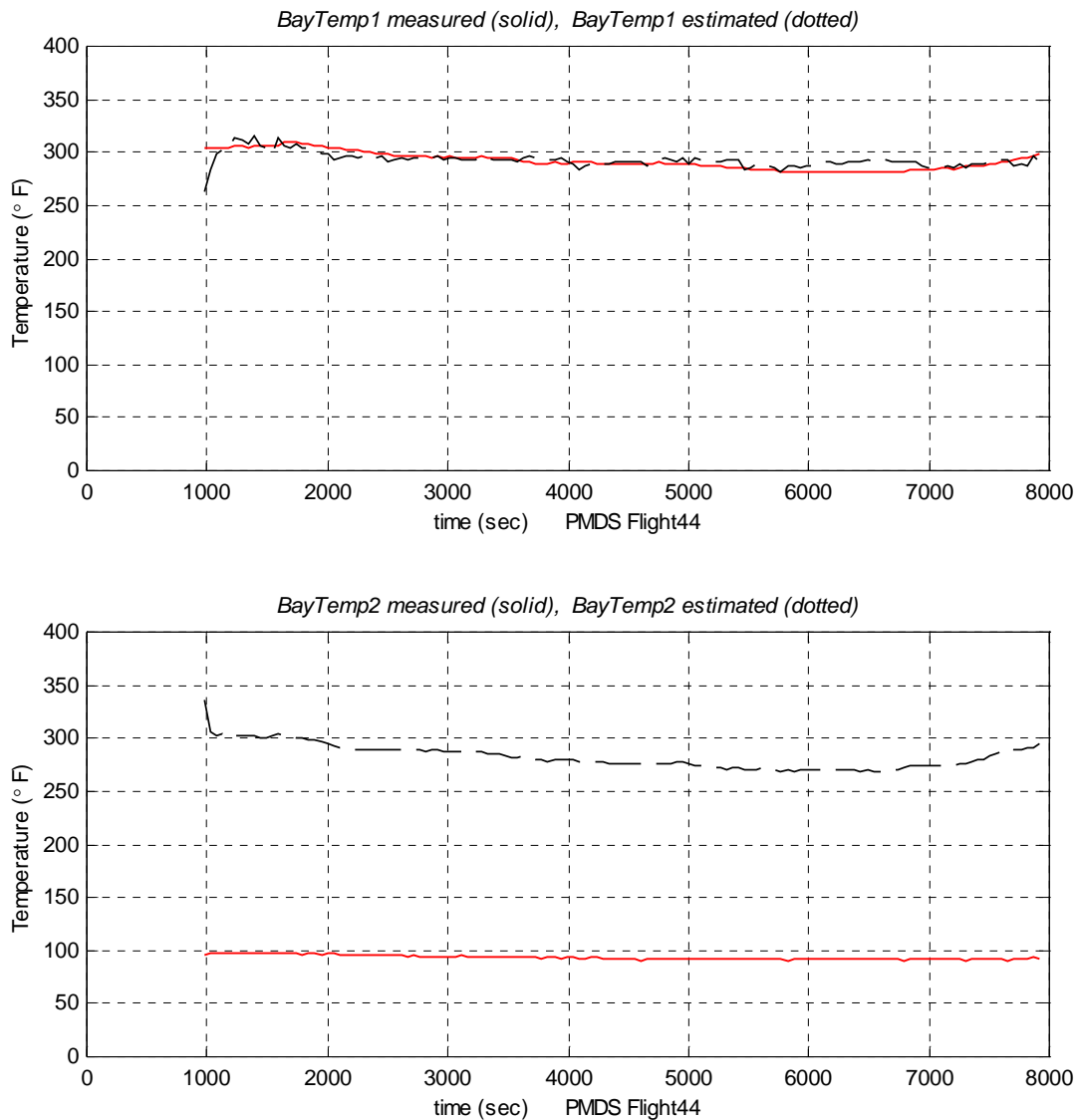
**Figure 3-3. CHT2 Measured Data and Modeling Data**



**Figure 3-4. CHT Fault Model Results, Flight 41**



**Figure 3-5. CHT Fault Model Results, Flight 44**



**Figure 3-6. Engine Bay Temperature Sensor Fault Model Results, Flight 44**

### 3.5 Linear Dynamic Modeling

A linear dynamic model can be used to estimate the value of one or more sensors using a past history of measured data from other sensors. This type of model is capable of producing a good estimate of the sensor values of interest. These estimated values can then be compared to measured sensor data for monitoring and diagnostic purposes.

### 3.5.1 Linear Dynamic Modeling Approach

The linear dynamic model is constructed in conventional state-space form, with a state equation and an output equation. The form of the dynamic model is as follows:

$$\dot{x} = Ax + Bu$$

$$y = Cx + Du$$

where

$x \in R^n$  is the state vector

$u \in R^m$  is the input vector

$y \in R^p$  is the output vector

$A \in R^{n \times n}$ ,  $B \in R^{n \times m}$  are the system matrices

Given a set of measured inputs over a specified time period, this model can be used to estimate the values of all modeled system states and outputs. In our analysis work, we generally did not compute the  $C$  and  $D$  matrices. (In a commercialized system, some states could be ignored and various outputs of interest could be defined and computed for monitoring and diagnostic purposes.)

Three dynamic models were developed using measured data from Flights 41 and 44. To compare the models, each used the same set of inputs and state variables. Any sensors that had simulated or intermittent faults were not used in the model.

Analysis of the flight test data showed that both test flights exhibited nonlinear characteristics (i.e., engine RPM was a nonlinear function of SLPC setting). For this reason, the dynamic models were linearized over the selected operating range. The resulting models were linearized about a nominal “trim” condition in the following form:

$$(\dot{x} - \dot{x}_{\text{trim}}) = A(x - x_{\text{trim}}) + B(u - u_{\text{trim}})$$

Since the  $\dot{x}_{\text{trim}}$  term is assumed to be zero, the above expression can be rewritten as

$$\dot{x} = Ax + Bu - (Ax_{\text{trim}} + Bu_{\text{trim}})$$

The last term,  $(Ax_{\text{trim}} + Bu_{\text{trim}})$ , can be approximated by an additional “bias” input (set to 1.0 in the  $u$  vector), thereby giving the conventional form:

$$\dot{x} = Ax + Bu$$



The model was computed using a least-squares approximation to measured data. The form of the model is

$$\dot{X} = [A, B] * [X; U]$$

where

$k$  = number of samples (period of time) over which the estimate is desired

$$\begin{aligned} \dot{X} &= [\dot{x}(t_1), \dots, \dot{x}(t_k)] \in R^{n \times k} \\ [X; U] &= \begin{bmatrix} x(t_1), & \dots & , x(t_k) \\ u(t_1), & \dots & , u(t_k) \end{bmatrix} \in R^{(n+m) \times k} \end{aligned}$$

The system matrices  $[A, B]$  were computed as follows:

$$[A, B] = \dot{X} [X; U]^T \left( [X; U] [X; U]^T \right)^{-1}$$

### 3.5.2 Model Data Preparation

The linear dynamic models were developed using the data collected in the two flight tests. The input data for the models was prepared as follows:

1. Measured data from Flight 41 and Flight 44 were placed into data files; this data was described in subsection 3.3.
2. Other pertinent data not collected automatically by the PMDS was added to the data files manually. This manually added data included the SLPC settings and cowl flap data collected from the flight test engineer's reports.

### 3.5.3 Linear Dynamic Model Development

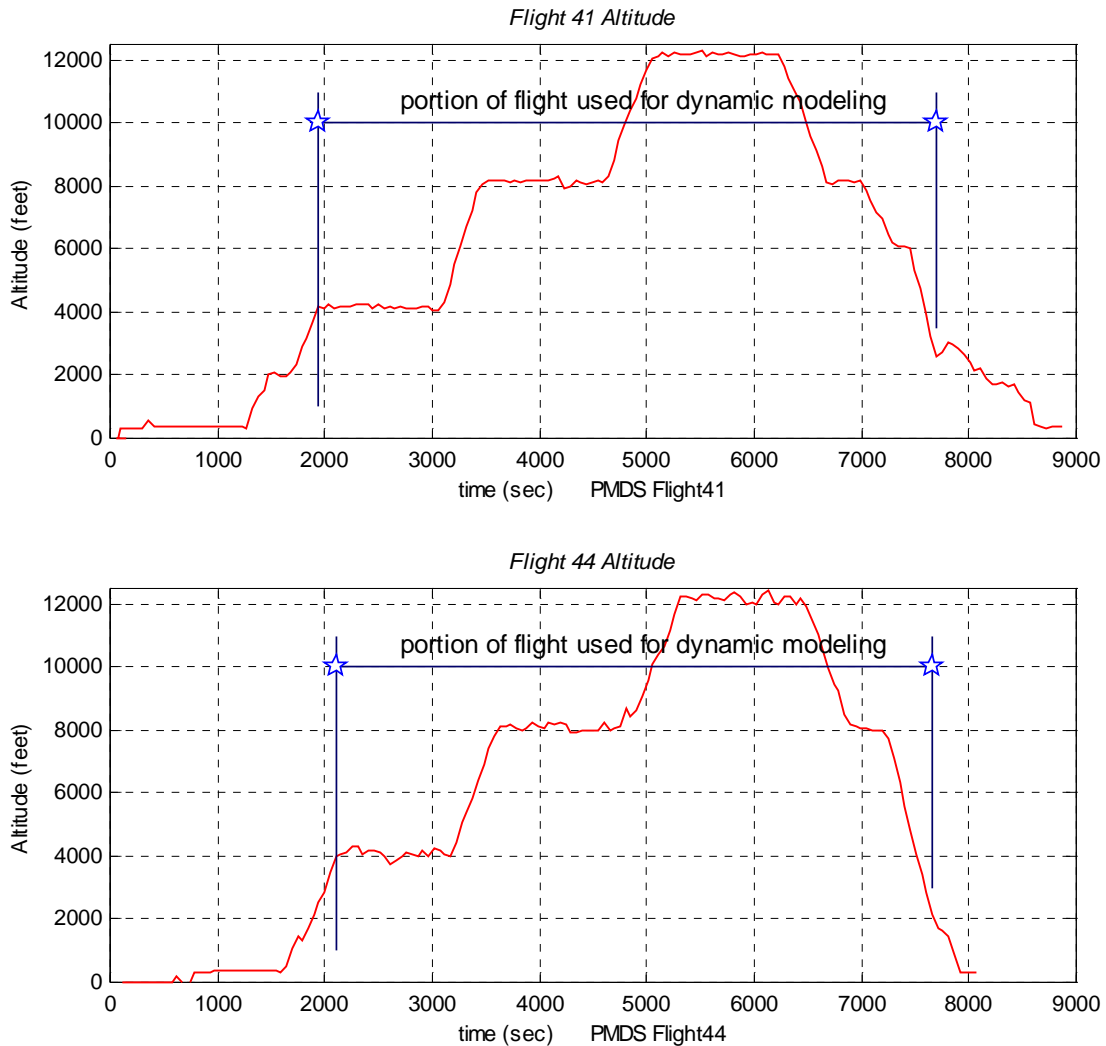
Three linear dynamic models were developed using the technical approach described above. These models are as follows:

- Model 41 created from Flight 41 test data
- Model 44 created from Flight 44 test data
- Model 4144 created from a combination of Flight 41 and Flight 44 test data

These models were developed from the flight test data as described above. The portions of the flight tests used in developing the models are shown in Figure 3-7.

The flight test data (inputs and states) used in developing the models is shown in Table 3-2.

Altitude slope, or rate of change, was used as a substitute for elevator setting (i.e., to indicate the load on the engine). This was done because we had no way to collect elevator position or pilot



**Figure 3-7. Linear Dynamic Model Data**

commands electronically. Similarly, we did not have a means to electronically collect airspeed data, and that data was not collected manually during the flights. However, some of the other sensor data effectively serves the same purpose (i.e., a combination of altitude, altitude rate, and SLPC setting).

Pressure altitude and air charge temperature inputs track other inputs either directly or inversely, but were used in order to produce a better model.

### 3.5.4 Linear Modeling Results

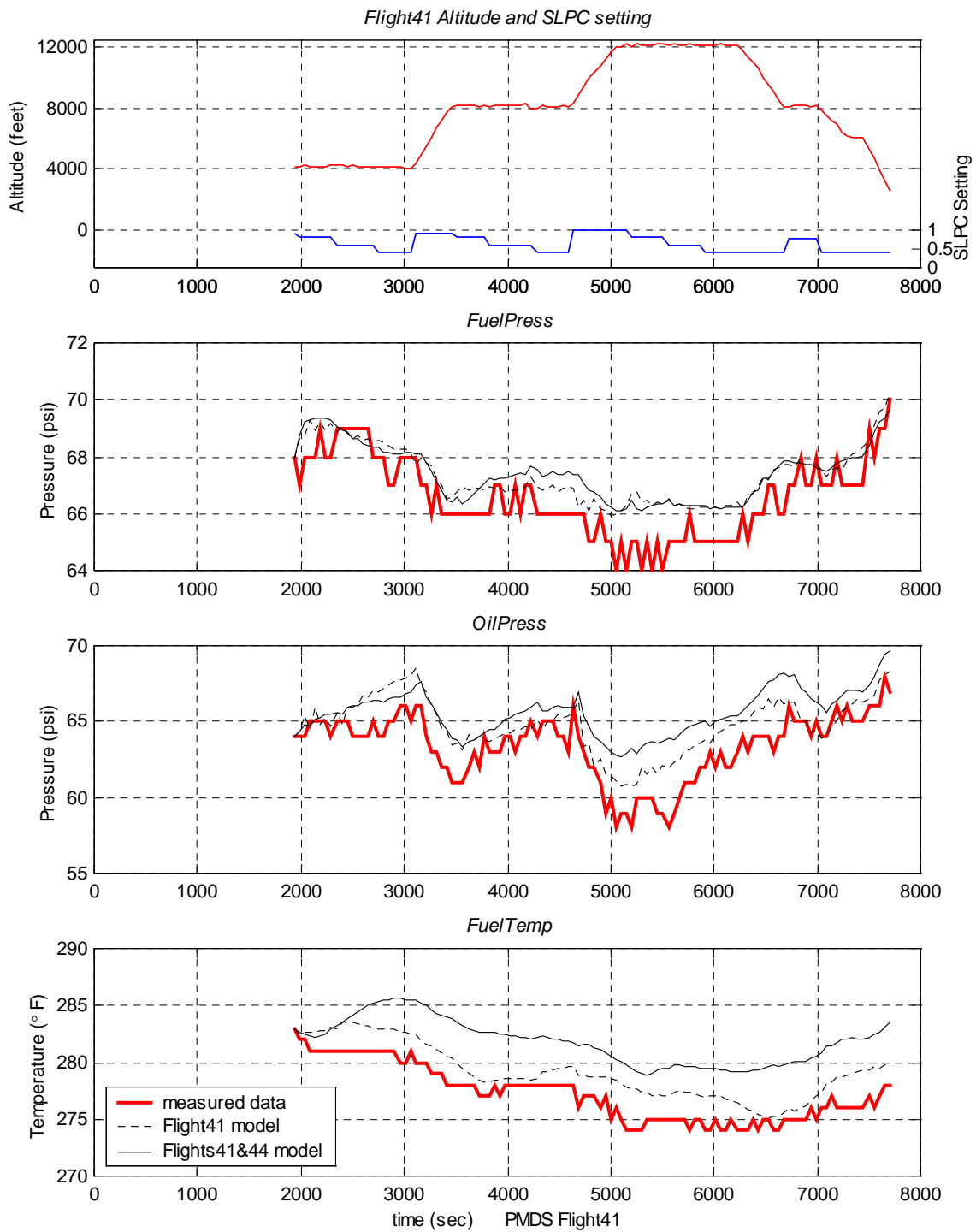
The above linear models were used to simulate a flight using initial conditions taken from either Flight 41 or Flight 44. Simulated results were compared with actual measured flight test data. The results are shown in Figures 3-8 and 3-9. The plots in Figures 3-8 and 3-9 were prepared as described in Table 3-3.

**Table 3-2. Linear Model Data**

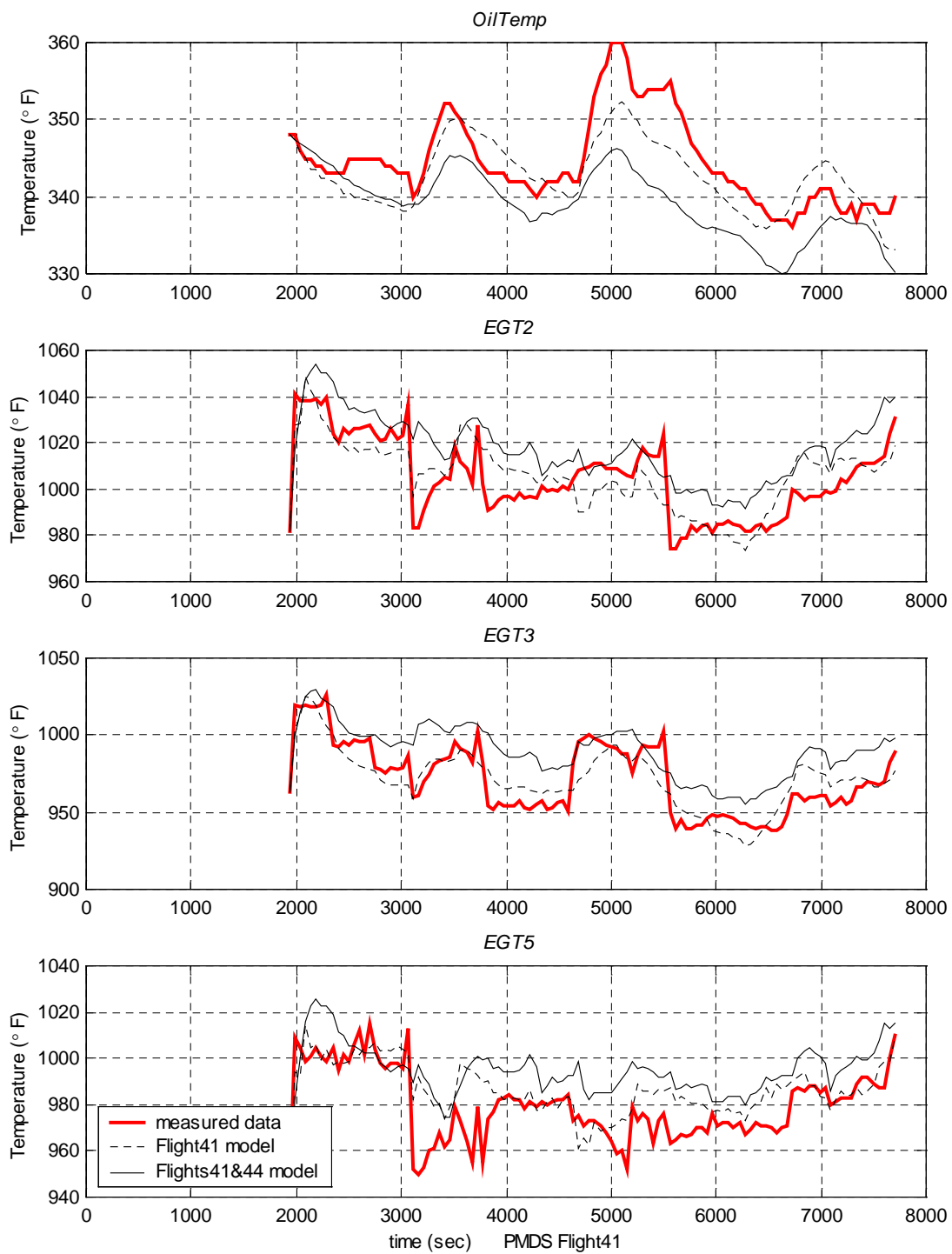
Inputs	States	Not Used	Bad Sensors
Altitude	Fuel Pressure	Kollsman	Bay 1 Temperature
Altitude Slope	Oil Pressure	Bay 2 Temperature	EGT1
Outside Air Temp	Fuel Temperature		EGT4
SLPC	Oil Temperature		CHT2
Cowl Flaps	EGT2		
Air Charge Temp	EGT3		
Pressure Altitude	EGT5		
(Dummy Input)	EGT6		
	CHT1		
	CHT3		
	CHT4		
	CHT5		
	CHT6		
	Manifold Air Pressure		
	Engine RPM		

**Table 3-3. Linear Model Plots**

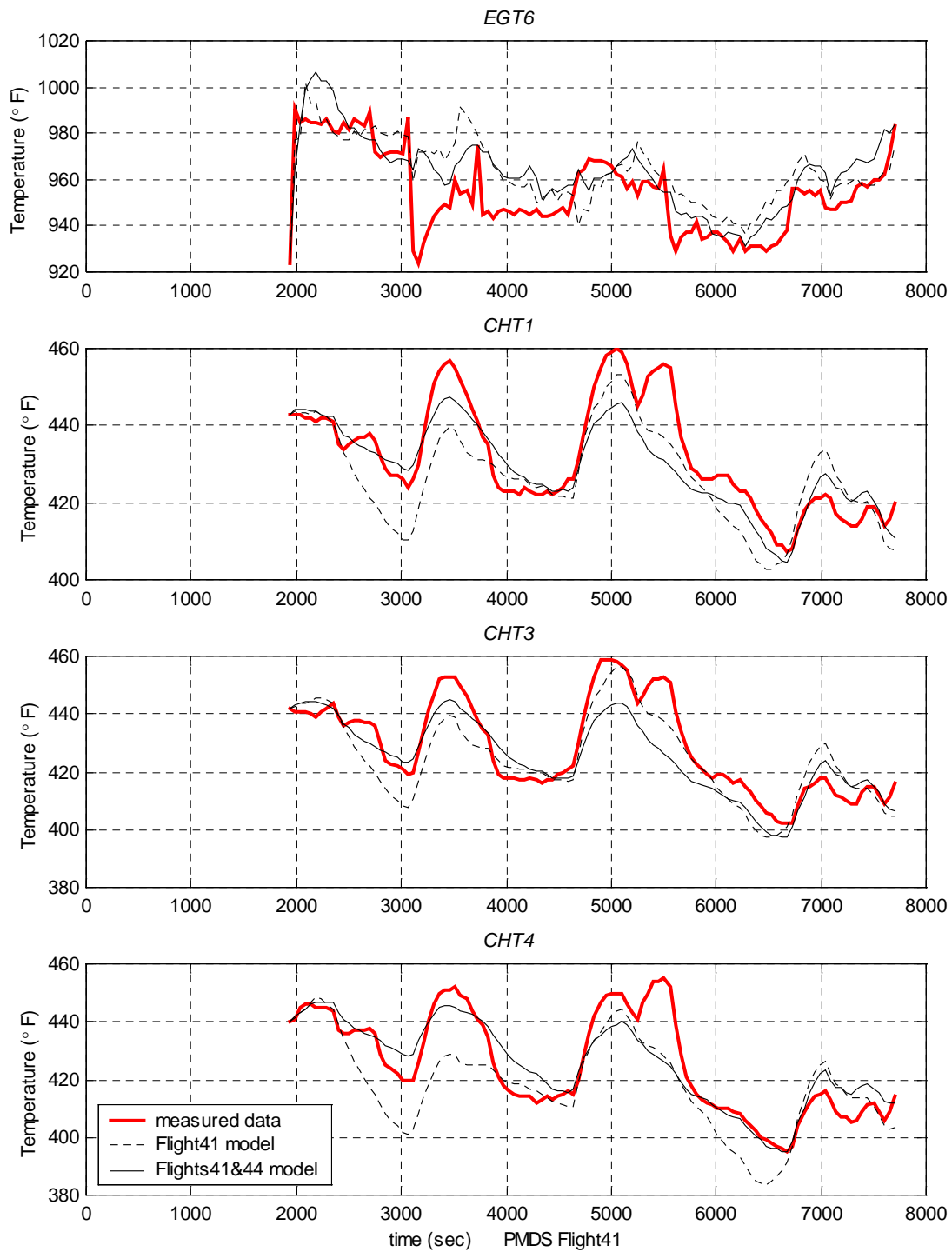
Figure	Flight	Models Used	Plotted Data
3-8	41	41 and 4144	Measured data: heavy line Linear Model 41 data: dashed Linear Model 4144 data: solid
3-9	44	44 and 4144	Measured data: heavy line Linear Model 44 data: dashed Linear Model 4144 data: solid



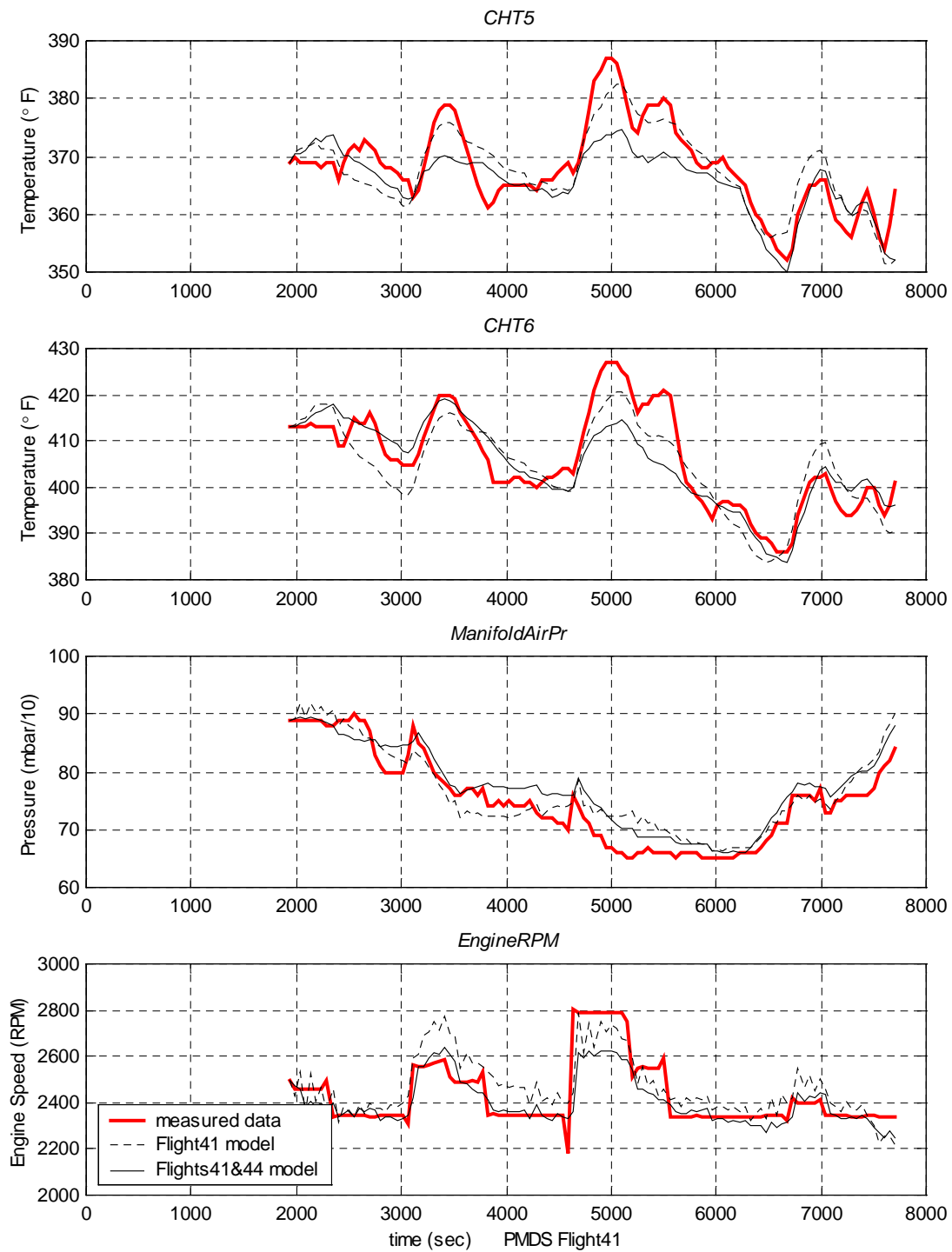
**Figure 3-8. Linear Dynamic Model Results, Flight 41**



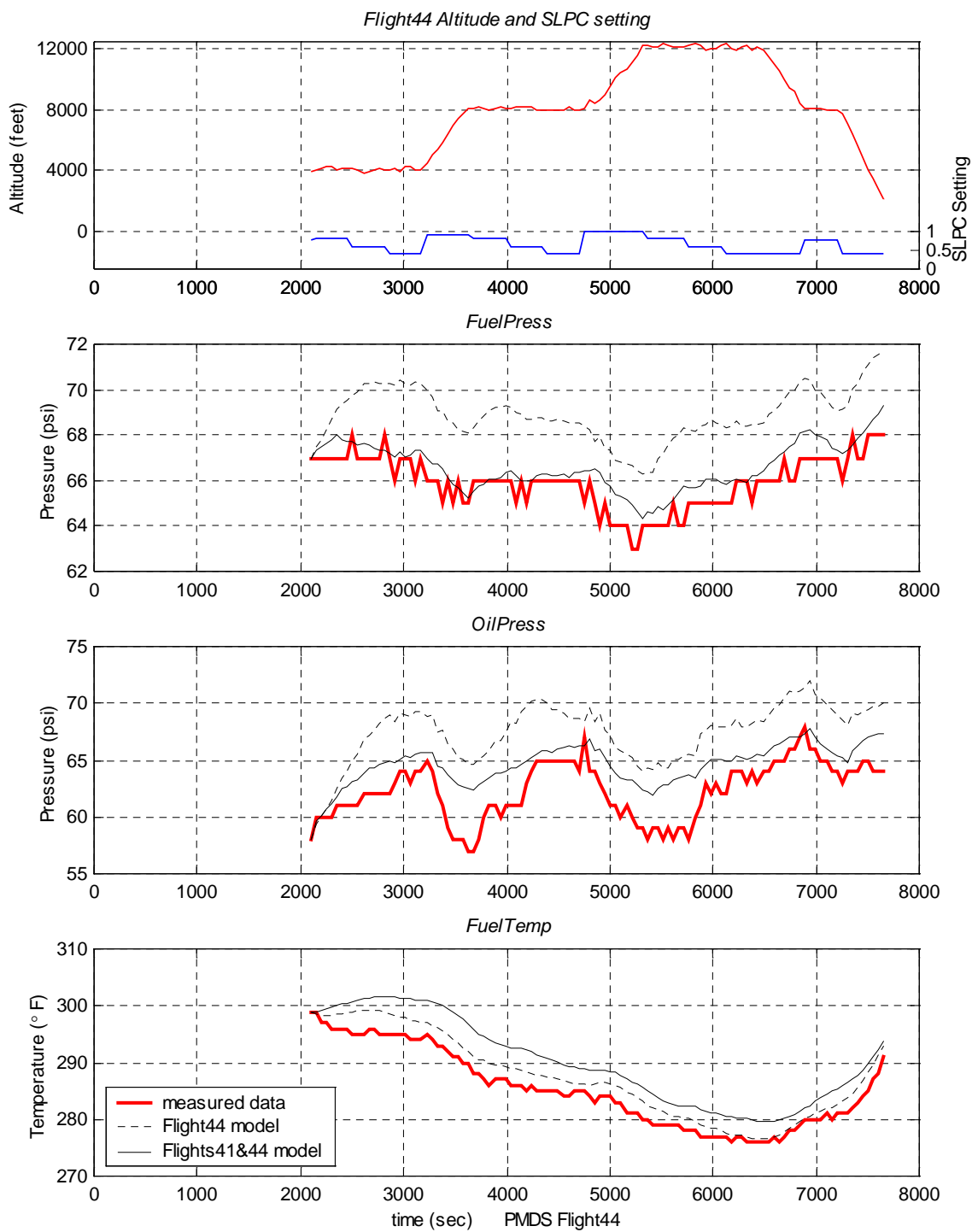
**Figure 3-8. Linear Dynamic Model Results, Flight 41 (continued)**



**Figure 3-8. Linear Dynamic Model Results, Flight 41 (continued)**

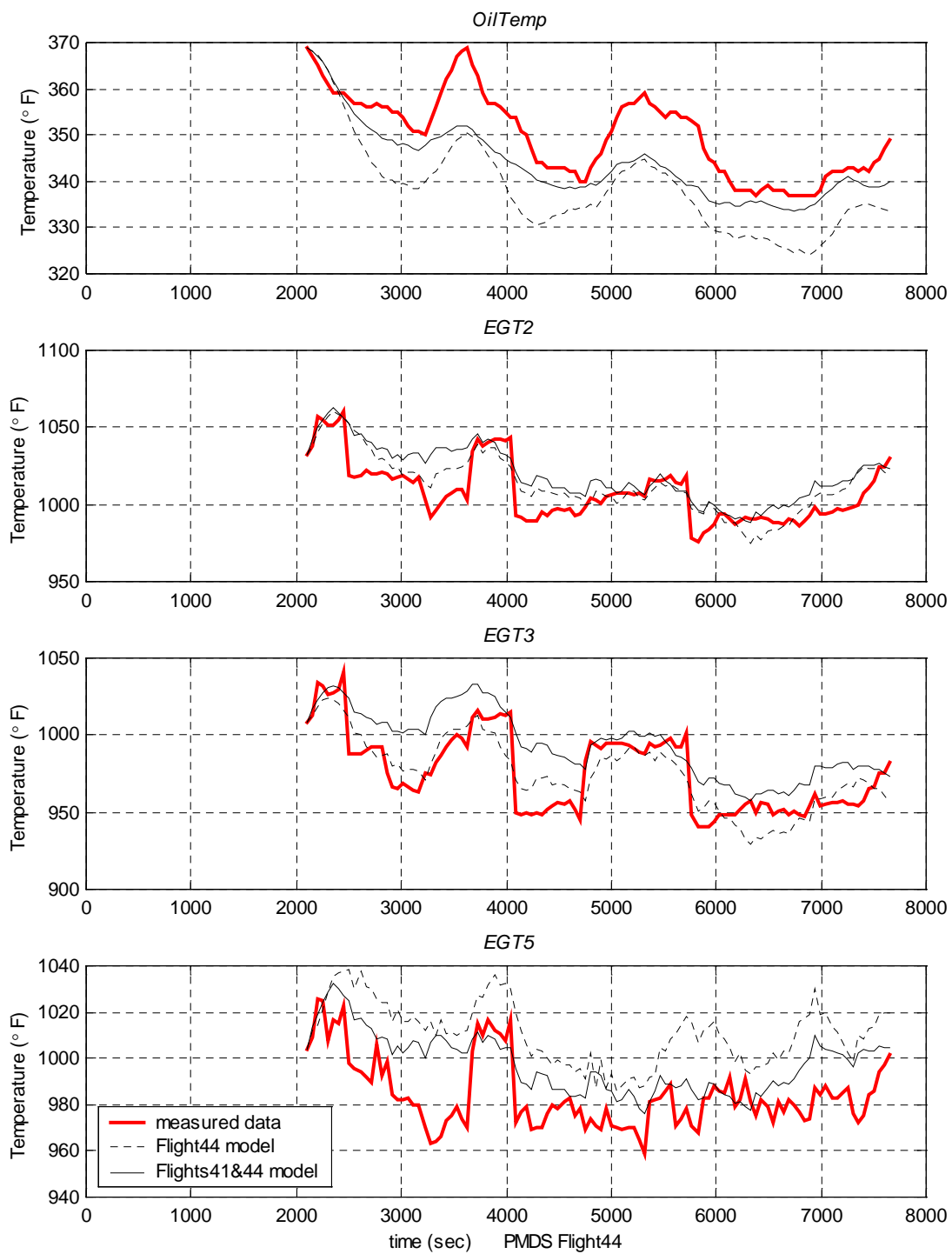


**Figure 3-8. Linear Dynamic Model Results, Flight 41 (concluded)**

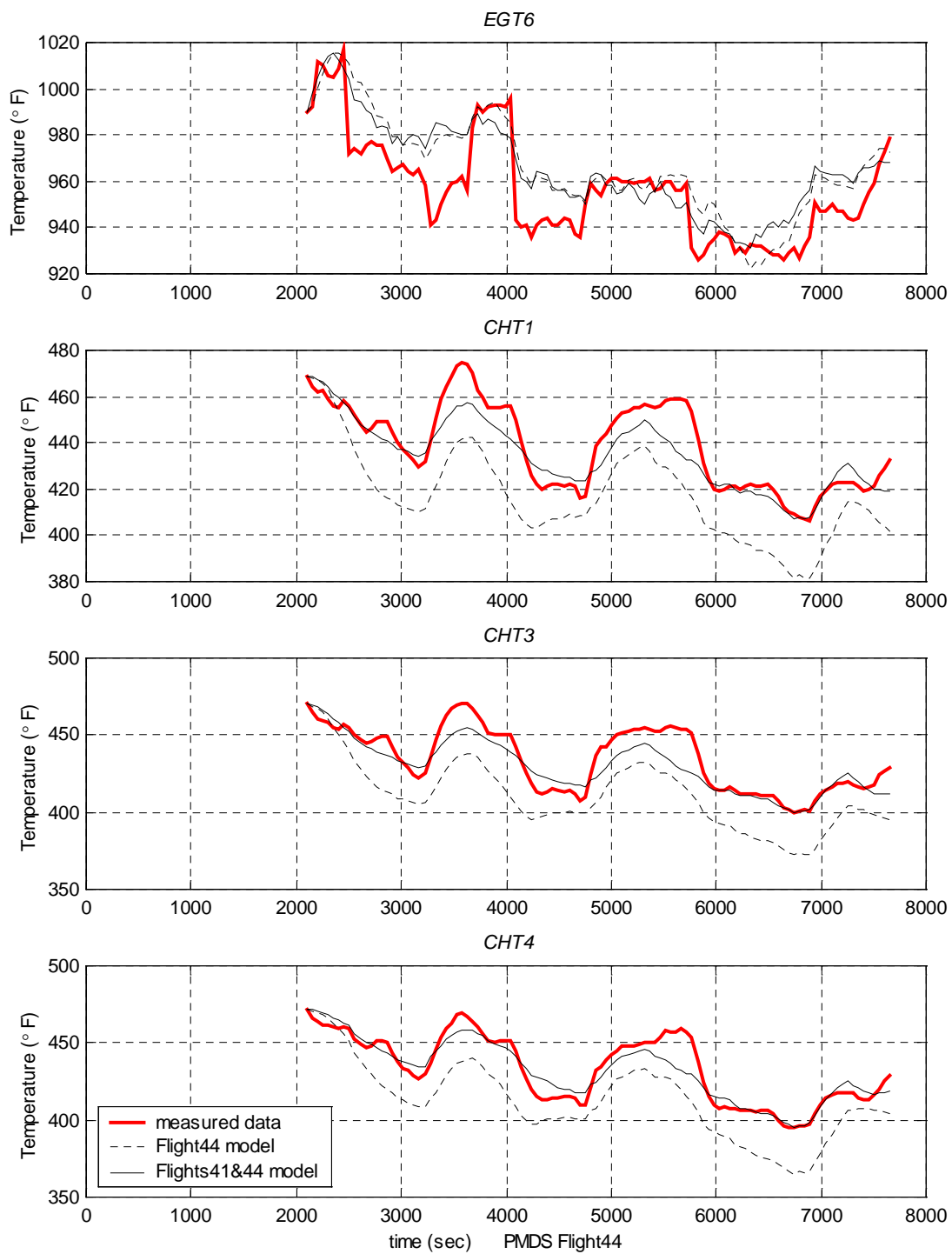


**Figure 3-9. Linear Dynamic Model Results, Flight 44**

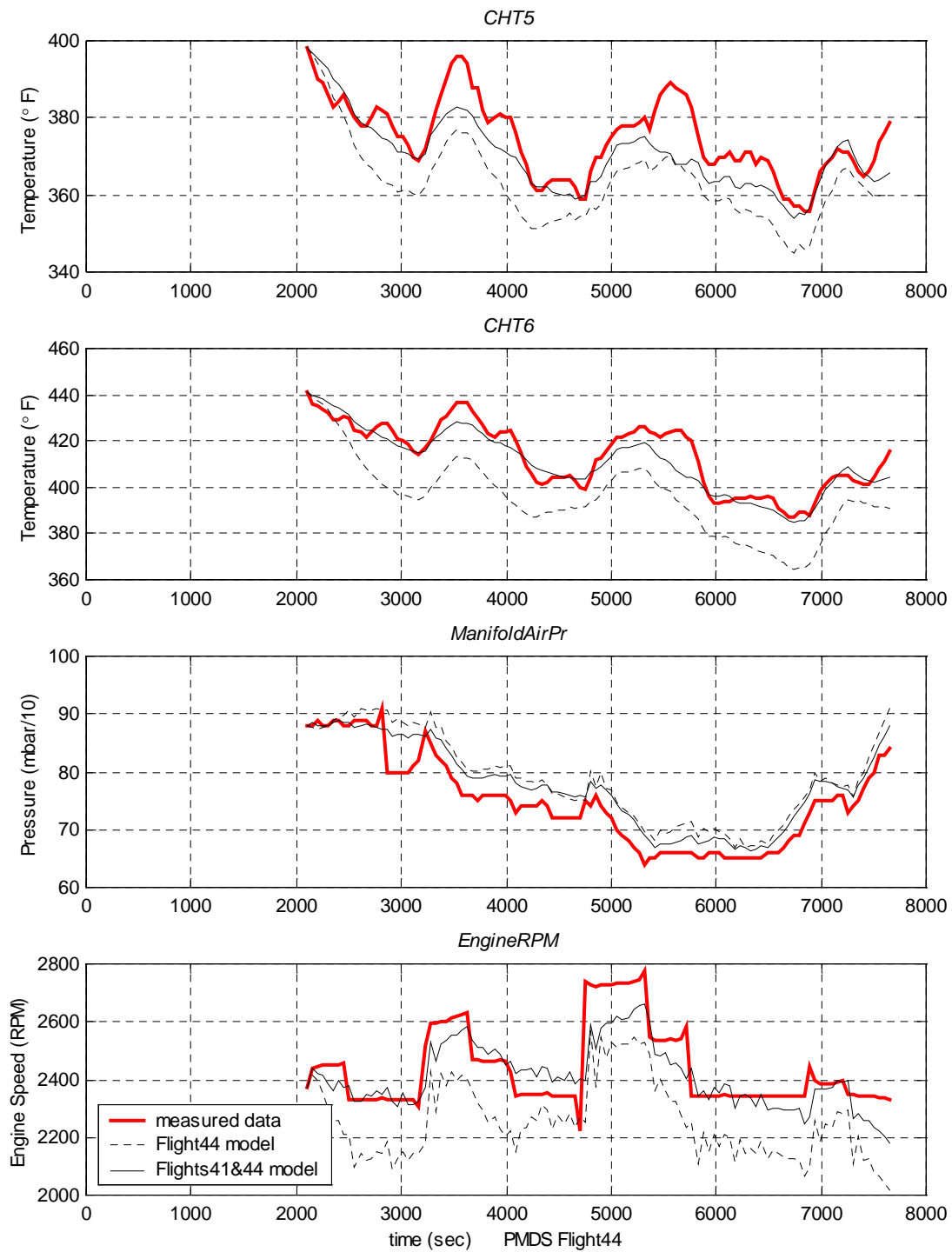




**Figure 3-9. Linear Dynamic Model Results, Flight 44 (continued)**



**Figure 3-9. Linear Dynamic Model Results, Flight 44 (continued)**



**Figure 3-9. Linear Dynamic Model Results, Flight 44 (concluded)**

### 3.6 Nonlinear Dynamic Modeling

In reviewing the flight test data, it was observed that there existed a nonlinear relationship between the engine RPM and the SLPC setting. As the SLPC setting was set to values just below 0.6, there was little change in engine RPM. Presumably in this region, the FADEC is adjusting the propeller pitch to satisfy the pilot's power level command. We were unable to acquire technical information about the FADEC control laws to verify this assumption. However, this measured characteristic of the SLPC and engine RPM response provides an opportunity to create a piecewise linear model consisting of two operating regions:

- Region 1: SLPC values greater than or equal to 0.6
- Region 2: SLPC values less than 0.6

#### 3.6.1 Nonlinear Dynamic Modeling Approach

The nonlinear dynamic model is constructed in state-space form. The form of the model is as follows:

$$\begin{aligned}\dot{x} &= Ax + Bu + b * \min(0, SLPC - 0.6) \\ y &= Cx + Du\end{aligned}$$

where

$x$  = state vector

$u$  = input vector

$y$  = output vector

$A, B, C, D$  = linear system matrices for Region 1 ( $SLPC > 0.6$ )

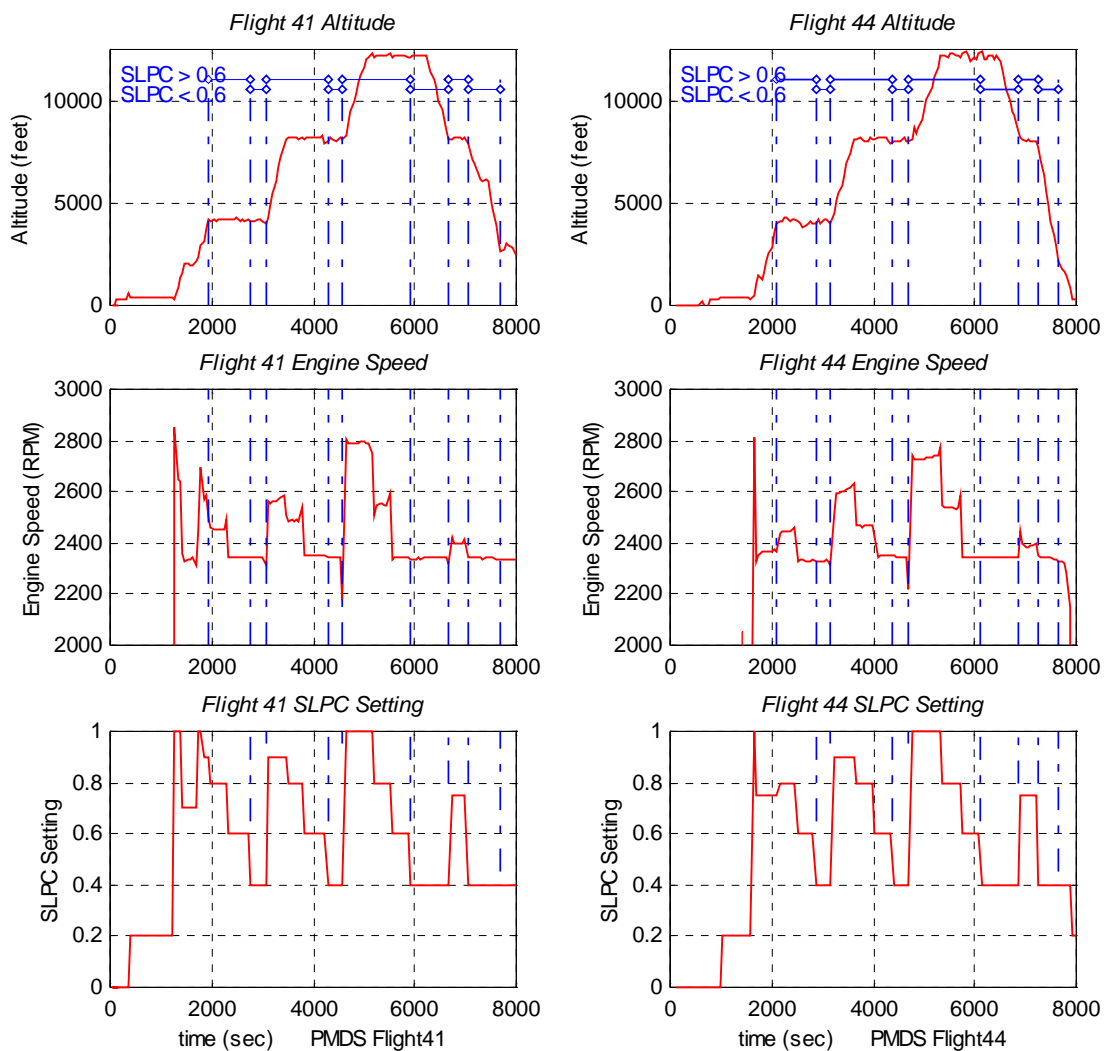
$b$  = vector of coefficients used to adjust the response for Region 2 ( $SLPC < 0.6$ )

The nonlinear model was developed using the following procedure:

1. Using measured data from Region 1, system matrices  $A$ ,  $B$ ,  $C$ , and  $D$  were computed using the method developed for the earlier linear dynamic models.
2. Using measured data from Region 2 and system matrices  $A$ ,  $B$ ,  $C$ , and  $D$ , the  $b$  vector was computed using a least-squares approximation.

#### 3.6.2 Model Data Preparation

The nonlinear dynamic models were developed using the data collected in the two flight tests. The flight test data files prepared earlier for the linear modeling were reused for this work. Region 1 and Region 2 of each flight were determined based on SLPC setting, as shown in Figure 3-10.



**Figure 3-10. Nonlinear Dynamic Model Data**

### 3.6.3 Nonlinear Dynamic Model Development

Three nonlinear dynamic models were developed using the technical approach described above. These models are as follows:

- NL Model 41 created from Flight 41 test data
- NL Model 44 created from Flight 44 test data
- NL Model 4144 created from a combination of Flight 41 and Flight 44 test data

The flight test data used in developing these models is shown in Table 3-4. Since the partitioning of the flight test data into two regions resulted in fewer data points in each region, we were forced to reduce the number of inputs and state variables in order to achieve models that were stable.

**Table 3-4. Nonlinear Model Data**

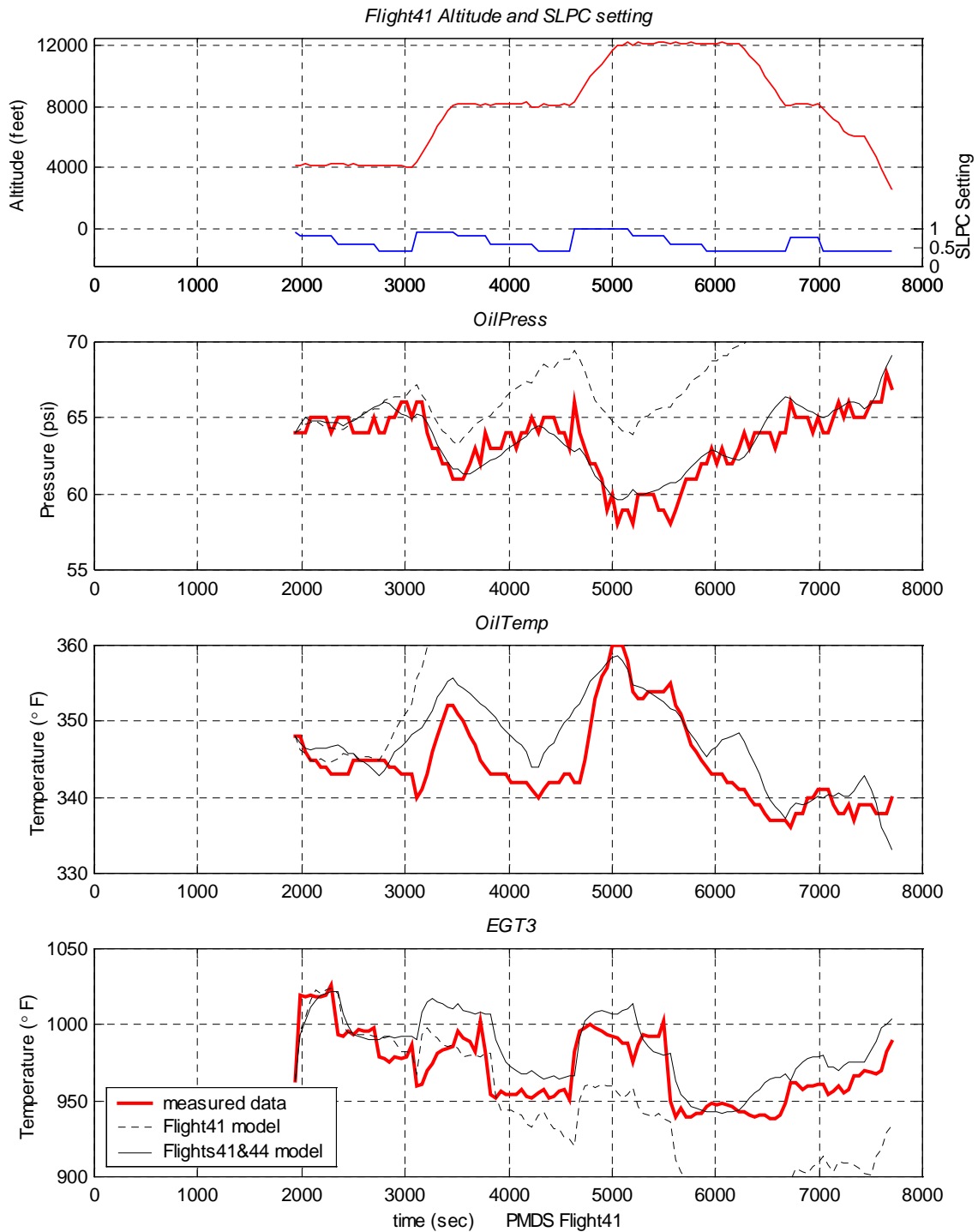
Inputs	States	Not Used	Bad Sensors
Altitude	Oil Pressure	Kollsman	Bay 1 Temperature
Altitude Slope	Oil Temperature	Bay 2 Temperature	EGT1
Outside Air Temp	EGT3	Fuel Pressure	EGT4
SLPC	CHT3	Fuel Temperature	CHT2
Cowl Flaps	Engine RPM	EGT2	
(Dummy Input)		EGT5	
		EGT6	
		CHT1	
		CHT4	
		CHT5	
		CHT6	
		Manifold Air Pressure	
		Air Charge Temp	
		Pressure Altitude	

### 3.6.4 Nonlinear Modeling Results

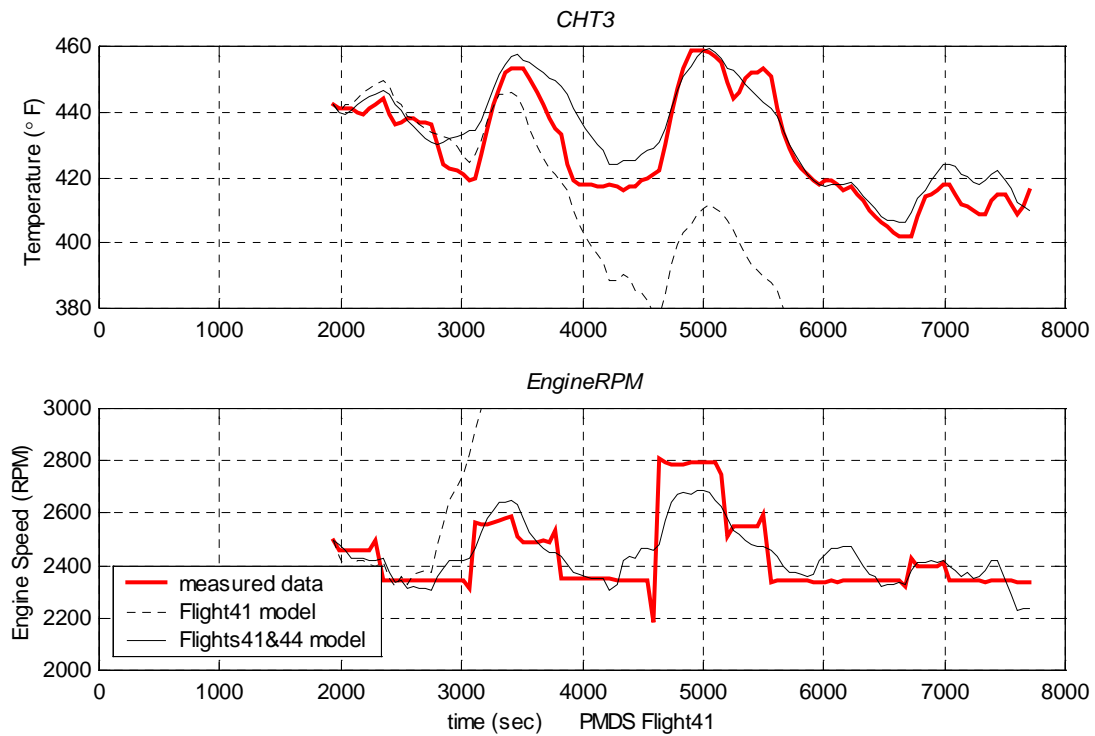
The above nonlinear models were used to simulate a flight using initial conditions taken from either Flight 41 or Flight 44. Simulated results were compared with actual measured flight test data. The results are shown in Figures 3-11 and 3-12. The plots in Figures 3-11 and 3-12 were prepared as described in Table 3-5.

**Table 3-5. Nonlinear Model Plots**

Figure	Flight	Models Used	Plotted Data
3-11	41	41 and 4144	Measured data: heavy line Nonlinear Model 41 data: dashed Nonlinear Model 4144 data: solid
3-12	44	44 and 4144	Measured data: heavy line Nonlinear Model 44 data: dashed Nonlinear Model 4144 data: solid

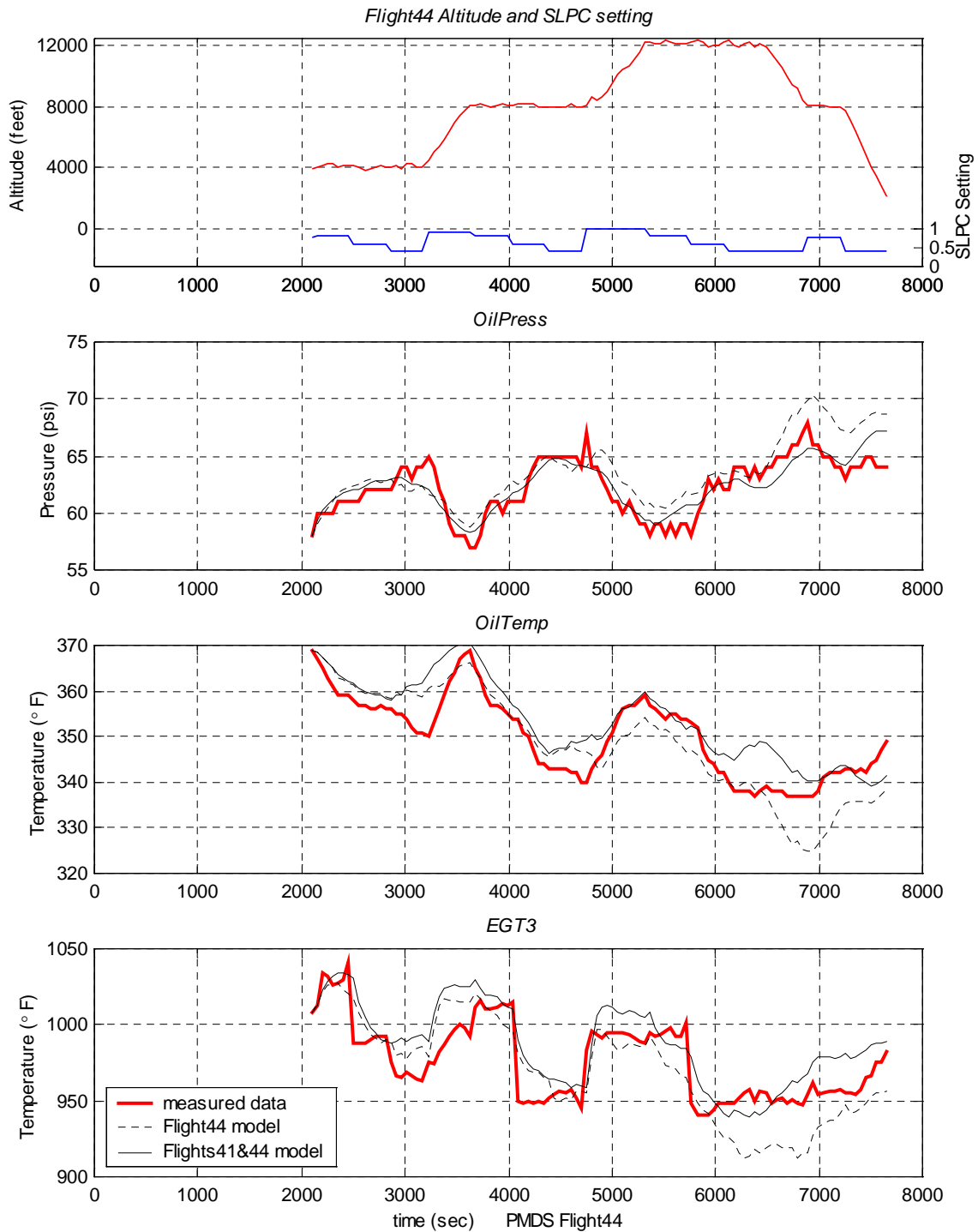


**Figure 3-11. Nonlinear Dynamic Model Results, Flight 41**

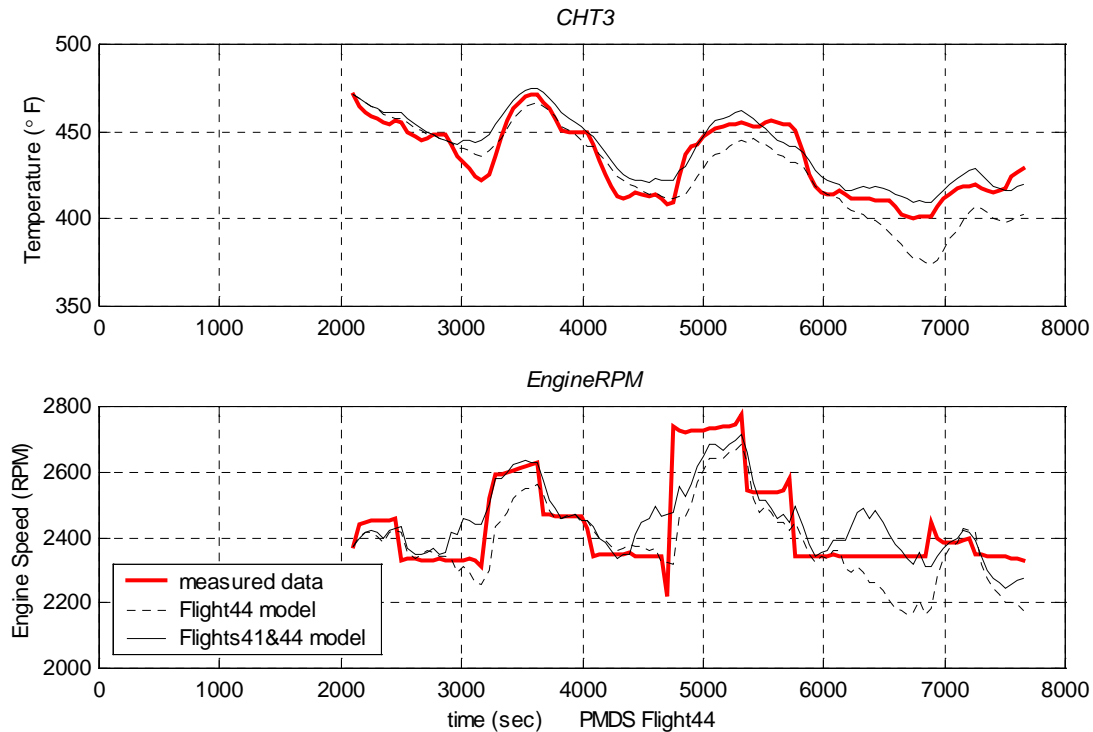


**Figure 3-11. Nonlinear Dynamic Model Results, Flight 41 (concluded)**





**Figure 3-12. Nonlinear Dynamic Model Results, Flight 44**



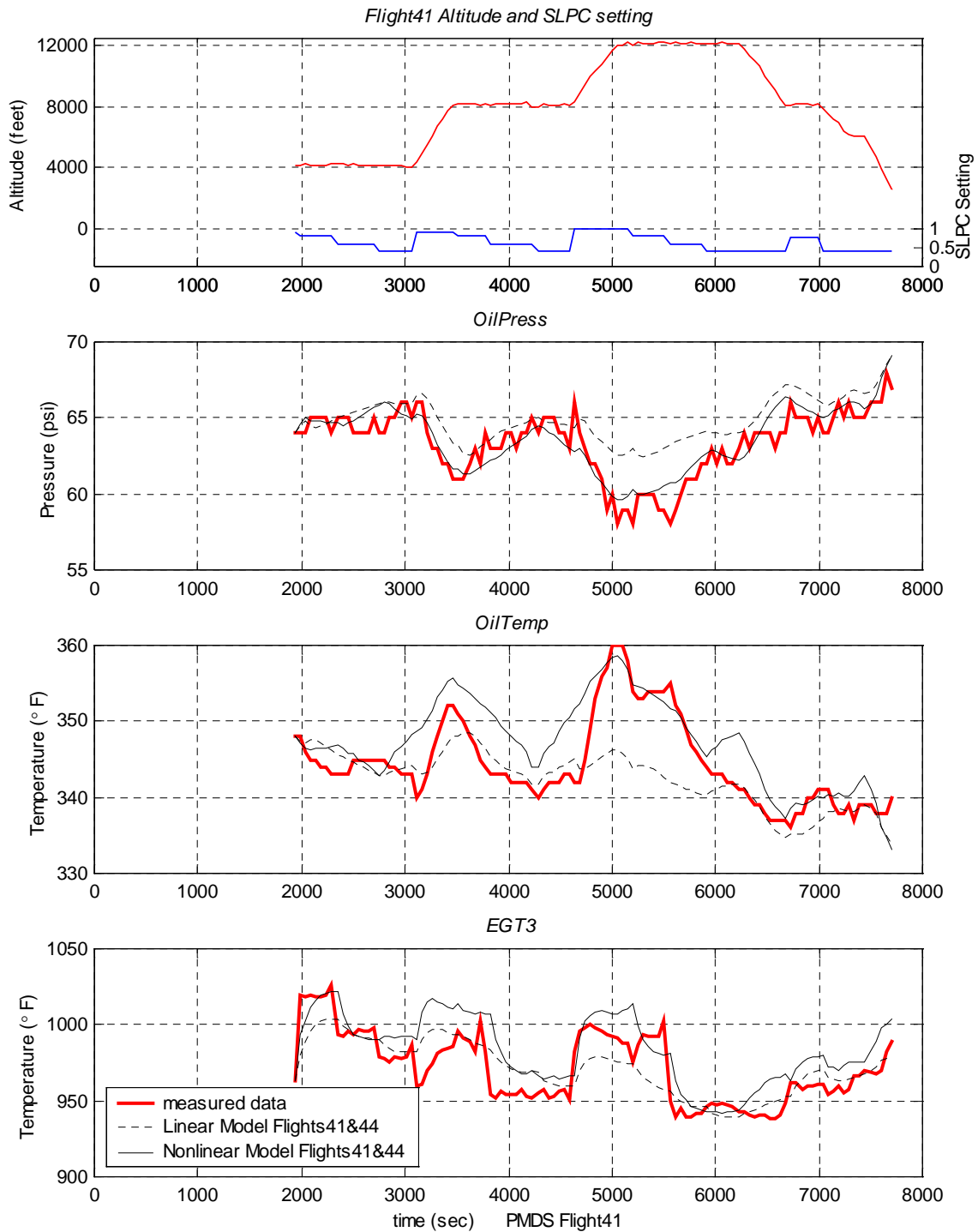
**Figure 3-12. Nonlinear Dynamic Model Results, Flight 44 (concluded)**

### 3.6.5 Comparison of Nonlinear and Linear Modeling Results

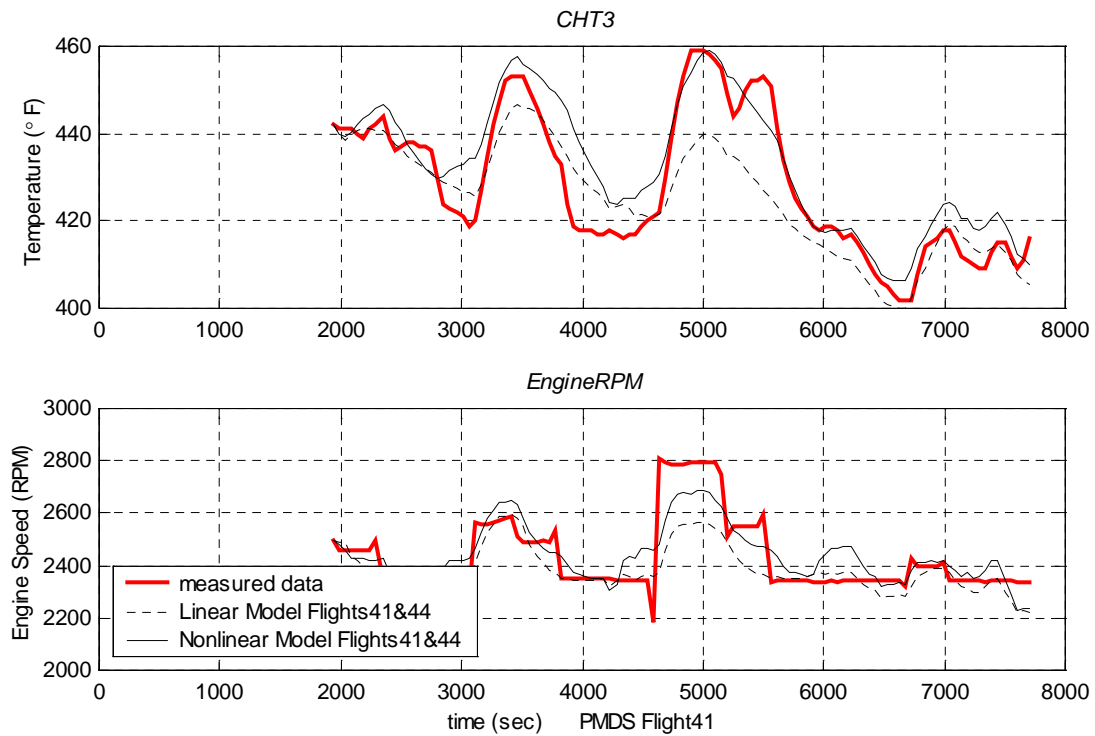
A simulation was performed to compare the quality of the linear and nonlinear modeling methods. The nonlinear model 4144 compared to a linear model 4144 using the same inputs and states (as listed in Table 3-4). The results are shown in Figures 3-13 and 3-14. The plots in Figures 3-13 and 3-14 were prepared as described in Table 3-6.

**Table 3-6 Comparison Plots**

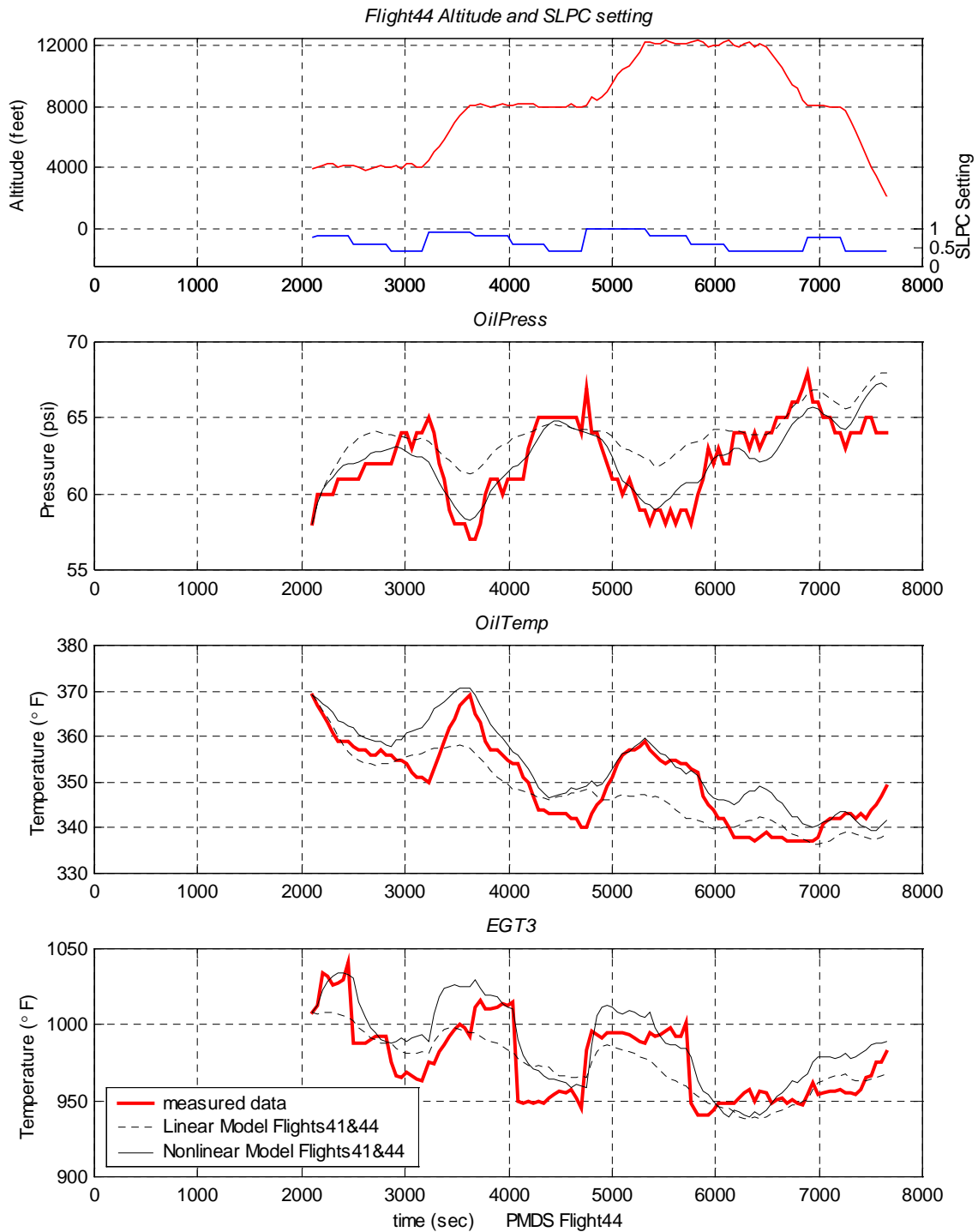
Figure	Flight	Models Used	Plotted Data
3-13	41	Nonlinear model vs. Linear model (modeled for combined Flights 41 and 44)	Measured data: heavy line Linear Model 4144 data: dashed Nonlinear Model 4144 data: solid
3-14	44	same as above	Measured data: heavy line Linear Model 4144 data: dashed Nonlinear Model 4144 data: solid



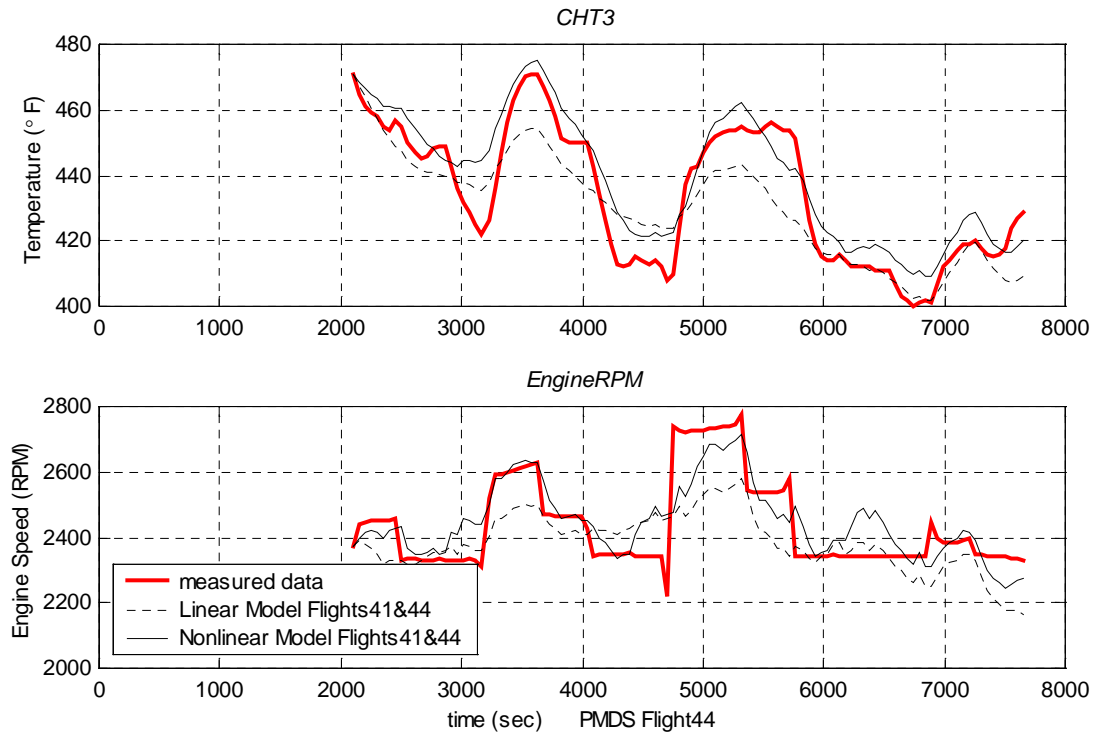
**Figure 3-13. Nonlinear vs. Linear Model Results, Flight 41**



**Figure 3-13. Nonlinear vs. Linear Model Results, Flight 41 (concluded)**



**Figure 3-14. Nonlinear vs. Linear Model Results, Flight 44**



**Figure 3-14. Nonlinear vs. Linear Model Results, Flight 44 (concluded)**

### 3.7 Data Analysis Conclusions

Conclusions drawn from the analysis of the flight test data and the resulting static correlation models and dynamic models are presented in the following subsections.

#### 3.7.1 Static Correlation Modeling Conclusions

Analysis of the static correlation modeling results provided the following observations:

- Disconnected sensors are clearly detectable using a static correlation model that compared sensor output values with expected values (see Figure 3-6).
- Intermittent sensor faults are also clearly detectable using a static correlation model (see Figure 3-2).
- Data from multiple flights can be combined to produce an improved static correlation model (see Figures 3-3, 3-4, and 3-5).
- Static correlation models worked well for detecting sensor faults.

### 3.7.2 Dynamic Modeling Conclusions

Analysis of the dynamic modeling results provided the following observations:

- Combining data from both test flights generally produced better models. This can be seen for the linear models in the last subplot in Figure 3-9 (comparing model 4144 and model 44).
- Nonlinear models were better than linear models for the same number of inputs and states. This can be seen by comparing the nonlinear and linear model results in the last subplot in Figure 3-13. The nonlinear models do, however, require more empirical data to generate.
- As expected, more data gives a better model. This can be seen by comparing the linear model in the last subplot of Figure 3-8 with that in the last subplot of Figure 3-13.
- Extending the dynamic model results from these two flight tests to future development (having many more flight tests) will produce much improved dynamic models. Combining the empirical modeling approach with a physics-based modeling approach also has the potential for improved accuracy.
- The two flight tests performed on this program demonstrate that dynamic models of the engine/aircraft can be produced using relatively simple and inexpensive instrumentation such as would be found in a commercializable on-board engine diagnostic system.
- Dynamic modeling has the potential to detect mechanical faults internal to the engine as well as sensor faults.

## Section 4. Technology Roadmap

The PMDS concept shows promise as a means to improve the pilot's awareness of the condition of the engine. This technology will require more development before it can be commercialized and broadly applied in the aviation marketplace. Honeywell has prepared an overview of the key technology areas that require further development. This information is presented for the purpose of guiding the direction of future development. The key technology development areas are

- Modeling and fault diagnosis algorithms development
- Hardware and software development
- System development

The following subsections discuss these three key areas and provide a roadmap for future development of each area.

### 4.1 Modeling and Fault Diagnosis Algorithms Development

The PMDS concept employs model-based diagnostic technology. Mathematical models for the engine, sensors, and related equipment are created from first-principles analysis and from empirical data collected from flight and ground-based testing. These models are used to detect faults in the engine, sensors, and related equipment. Fault information is used to make failure diagnoses. This process (for empirical models) is shown in Figure 4-1.

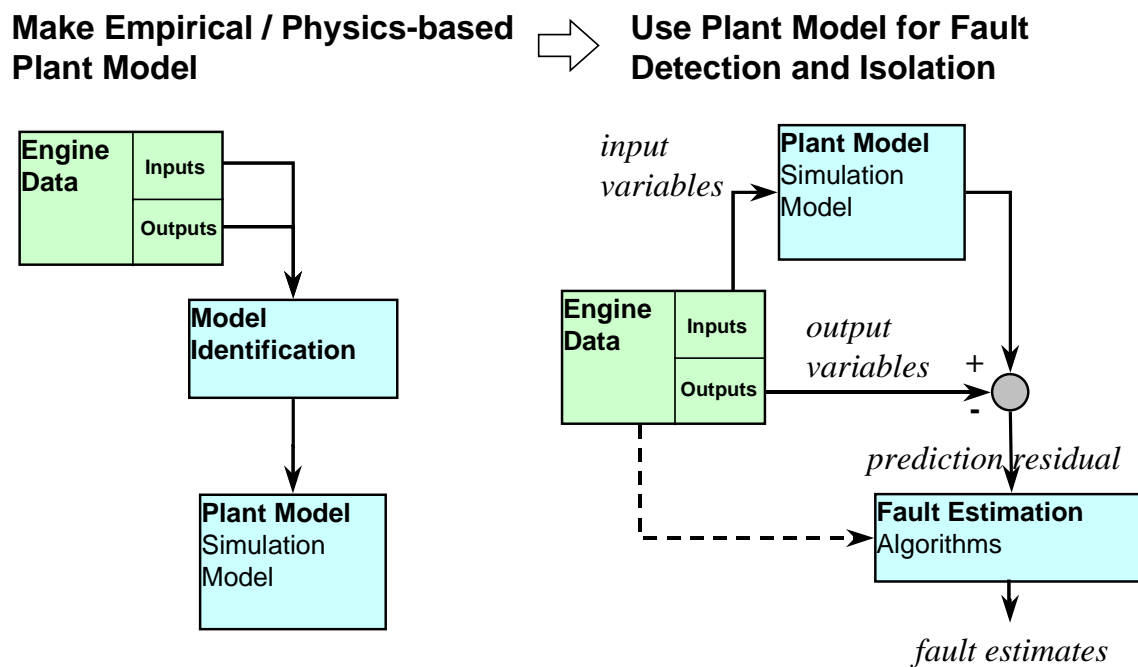


Figure 4-1. Modeling and Fault Detection



### 4.1.1 Background

**Modeling**—System modeling for engines can be done in various ways. Empirical modeling uses input/output data to derive a model of the engine. First-principles modeling uses physics and thermodynamics, etc. to derive a model of the engine. The pros and cons of each approach, as well as that of a combined approach, are discussed below.

The advantage of empirical models is that they can be very simple and general enough to apply to a variety of systems with little change from system to system. The disadvantage of empirical models is that they typically require many unknown coefficients that need to be identified using extensive empirical data. A general rule, based on the Cramer-Rao bounds, is that the number of required sample points is proportional to the square of the number of unknown coefficients. Given enough sensors and sample points, it is possible to evaluate all the coefficients, but if any system dynamics change, it may be necessary to start over. For example, given a system with  $x \in R^n$ , numerically differentiated state derivatives  $\dot{x} \in R^n$  and  $u \in R^m$ , coefficient matrices  $A \in R^{n \times n}$ ,  $B \in R^{n \times m}$ , and a linear empirical model of the form  $\dot{x} = [A, B] * [x; u]$ , the unknown coefficient matrices  $[A, B]$  can be evaluated using  $k$  samples of each of the  $n+m$  measurements of  $x$  and  $u$ . Let

$$F = [\dot{x}(t_1), \dots, \dot{x}(t_k)] \in R^{n \times k}$$

$$G = \begin{bmatrix} x(t_1), \dots, x(t_k) \\ u(t_1), \dots, u(t_k) \end{bmatrix} \in R^{(n+m) \times k}$$

Then  $F = [A, B] * G$  and  $[A, B] = F * G' * \text{inv}(G * G')$ .

Since the  $n+m$  coefficients in each row of  $[A, B]$  only affect the derivative of the corresponding state, those  $n+m$  parameters need to have  $k > (n+m)^2$  samples of that state derivative in order to satisfy the quadratic Cramer-Rao bounds. With  $n = 5$  and  $m = 5$ , that requires  $k > (5+5)^2 = 100$ . With  $n = 10$  and  $m = 5$ , we would need  $k > (10+5)^2 = 225$  samples. With the test flights used in this study, we had 110 good test points in each of the two flights, for a total of  $k = 220$ . This means that a single flight, with 110 samples, is enough to identify a five-state model, while a ten-state model would require the data from both flights together.

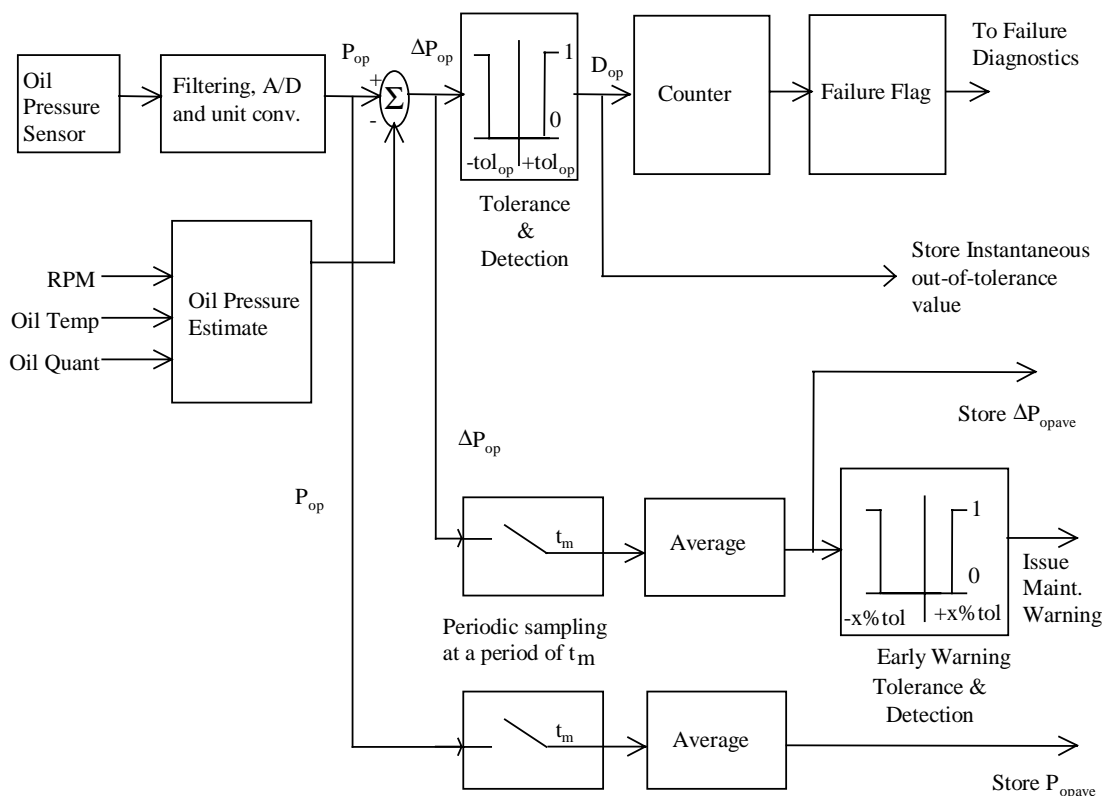
The advantages of a first-principles model include the ability to do “what if” experiments with the model and the ability to adapt the model to new untested situations. Another advantage of first-principles models is that they typically have fewer parameters than empirical models. It is possible to take advantage of the structure imposed by multiple time scale data and the natural separation of the system dynamics to create multiple subsystem models.

The parameters in a first-principles model are closely related to things that can be measured in isolation from the rest of the system. This makes it possible to combine the general structure of the first-principles model with empirical data to calibrate a few unknown parameter values. The disadvantage of first-principles models is that they take considerable time to derive and must be tailored to each specific application.

In our application, the only nonempirical information that we used was the nature of the nonlinearity in the SLPC. The SLPC input controls a combination of RPM and propeller pitch. The system behaves essentially like one linear system above  $\text{SLPC} = 0.6$ , and like another linear

system when SLPC is at lower levels. Rather than doubling the number of parameters, we kept the same  $A$  matrix and simply added one column to the  $B$  matrix, which multiplied the added input:  $\min(0, \text{SLPC} - 0.6)$ .

**Failure Detection and Fault Diagnosis**—During the AGATE program, a set of failure detection and fault diagnosis algorithms was developed. A set of engine models is used in computing an estimate of each of the various engine performance parameters needed for fault diagnosis. An example failure detection algorithm for oil pressure is shown in Figure 4-2.



**Figure 4-2. Oil Pressure Failure Detection and Warning Diagnosis**

The anticipated failures and the associated detection for each of the engine subsystems are arranged in a matrix of probable faults with the associated detection methods. An example fault diagnosis matrix related to oil pressure is shown in Table 4-1.

**Table 4-1. Oil Pressure Fault Modes and Corresponding Detection Tests**

Detection Test Methodology	Fault Mode					
	Loss of pressurized, filtered oil output to the oil cooler assembly	a) Low or no oil in sump	b) Failed/weak pressure regulator spring or worn pump elements	c) Seized oil pump or failed drive shaft	Loss of oil delta P sensor output to the FADEC	
					a) Electrical open or short within the sensor circuits	
					b) Delta P sensor physically separated from sensor port	
					(other faults . . .)	
					(other faults . . .)	
Difference between oil pressure estimate and oil pressure above detection threshold	◆		◆	◆		
Difference between oil delta P estimate and oil delta P above detection threshold			◆	◆	◆	◆
(Other tests . . .)						
(Other tests . . .)						

The current fault detection amounts to noting when a measured output deviates (by more than some threshold amount, for several samples) from the value of the model output, when the model is being fed the same inputs as the actual system. The current fault diagnosis assumes that the fault lies with the sensor whose value is deviating from the model prediction. A more detailed fault diagnosis would have to include engine faults as well as sensor faults.

An empirical way to calibrate an engine fault model would be to record sensed variables with and without each expected fault. Another option would be to derive a first-principles model of how each fault affects each sensor output. It would likely be necessary to combine the methods, using a first-principles model and using empirical data to calibrate the remaining unknown coefficients associated with the faults.

#### 4.1.2 Future Development

Professor Giorgio Rizzoni and Gary L. Parker at Ohio State University have developed a first-principles individual-cylinder model of an internal-combustion engine under the AGATE program. This model was implemented in Simulink. In future applications of our PMDS technology, we would like to use our existing flight data to evaluate some of the coefficients in such a first-principles model. For example, an empirical modification is often needed for the first-principles model of the relationship between throttle setting and manifold pressure. We

would also like to get empirical data for the most common types of faults experienced by general aviation engines.

A PMDS also would benefit greatly from additional inputs from outside the engine, such as aircraft speed, aileron, elevator, rudder, and flap settings, propeller pitch, etc. With a combination of a first-principles model and the availability of sufficient measurements of engine and aircraft variables, it will be much easier to achieve a good calibration of the model coefficients.

Future development of an engine monitoring and diagnostic system will require a prioritized list of engine faults to guide the diagnostic development work. Ideally, this technical information should come from an engine manufacturer. Data from more than one engine manufacturer would be even more helpful in advancing the technology.

While at present most general aviation operators are using gasoline-powered piston engines, current development programs in alternative-fuel engines, such as diesel engines, can offer a means to collect the engine fault information needed for the eventual development of an engine monitoring and diagnostic capability for these new engine technologies.

The application of vibration monitoring is an obvious area of interest in the subject of engine monitoring and diagnostics. Certain engine conditions such as bearing wear are best detected through vibration monitoring. The AGATE PMDS hardware was designed to optionally take information from a vibration sensor mounted directly on the engine, process the vibration data, and determine prognostics from that data. A university-led study performed under the Honeywell AGATE program has shown that, for piston engines, vibration monitoring can be used to detect engine conditions such as bearing wear. However, the team discovered that it was not sufficient to analyze the frequency spectrum of the vibration data as in traditional vibration monitoring methods, but rather to use a direct sample of the vibration time signal. This preliminary study used 5000 samples/second for one second of engine operation in order to detect engine conditions sufficient for prognostic prediction. Further research and development will be necessary before such optional vibration data can be used in a reliable fashion for engine prognostics.

The future development steps described above are shown in Figure 4-3. At this time, the scope of the work, the timing, the source of development funding, and the makeup of the development team are undefined. These planning issues will be addressed as the general aviation community continues to dialog about engine diagnostics technology and market needs.

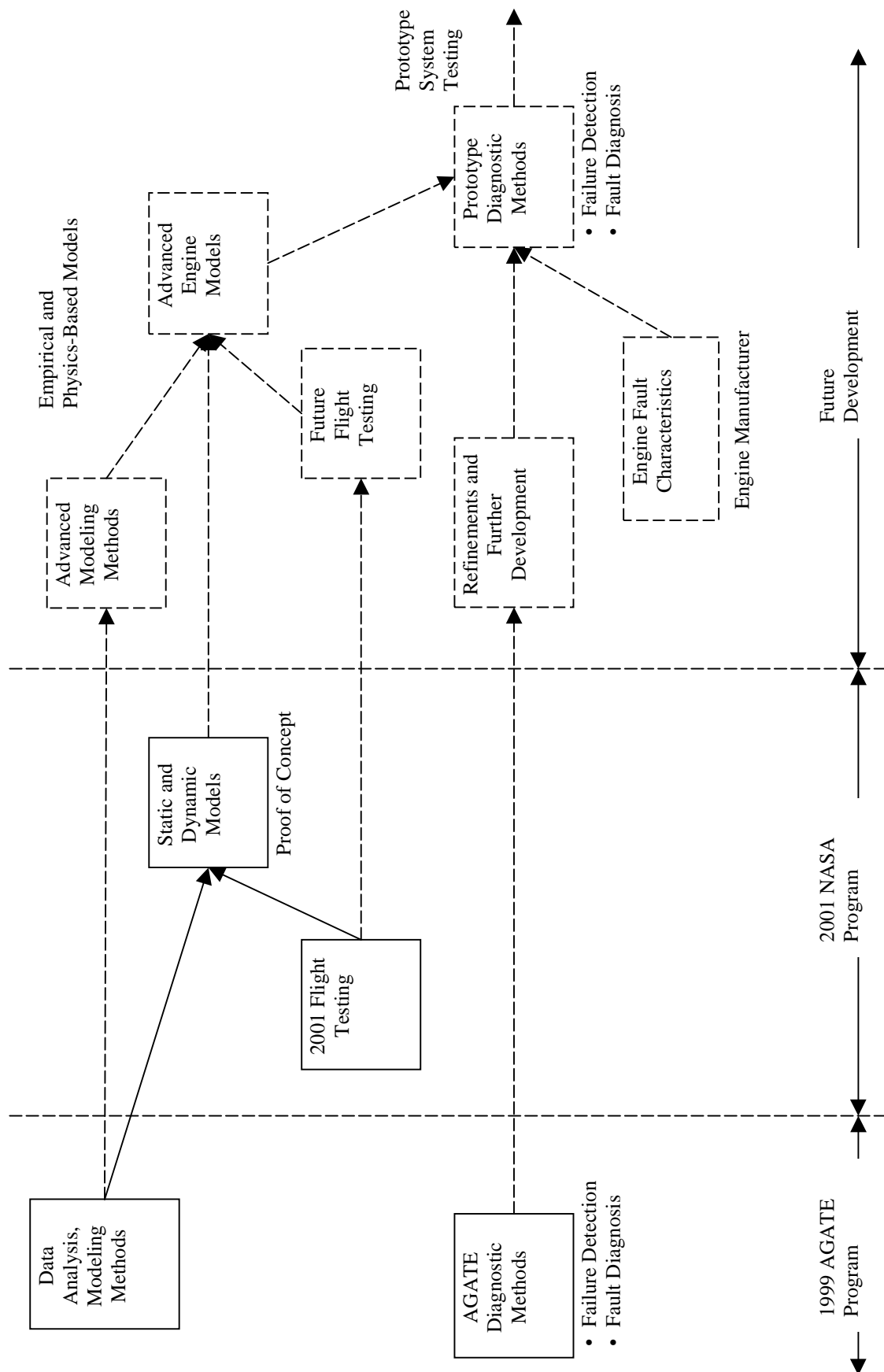


Figure 4-3. Modeling and Fault Diagnosis Algorithms Development Roadmap

## 4.2 Hardware and Software Development

During the AGATE program, Honeywell developed a prototype design for the PMDS. This hardware and software design will require additional development to bring it to a commercializable form. The following subsections describe the key hardware and software development areas ahead.

### 4.2.1 Background

The AGATE prototype system architecture concept is shown in Figure 4-4.

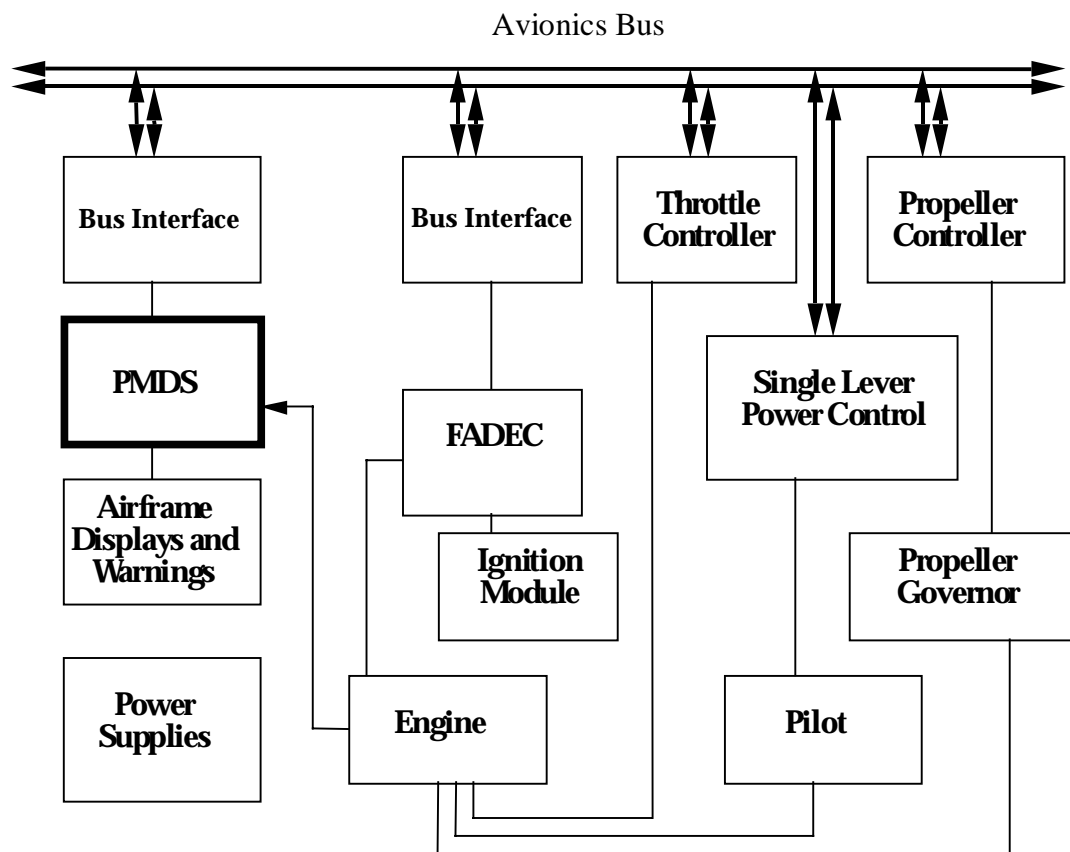
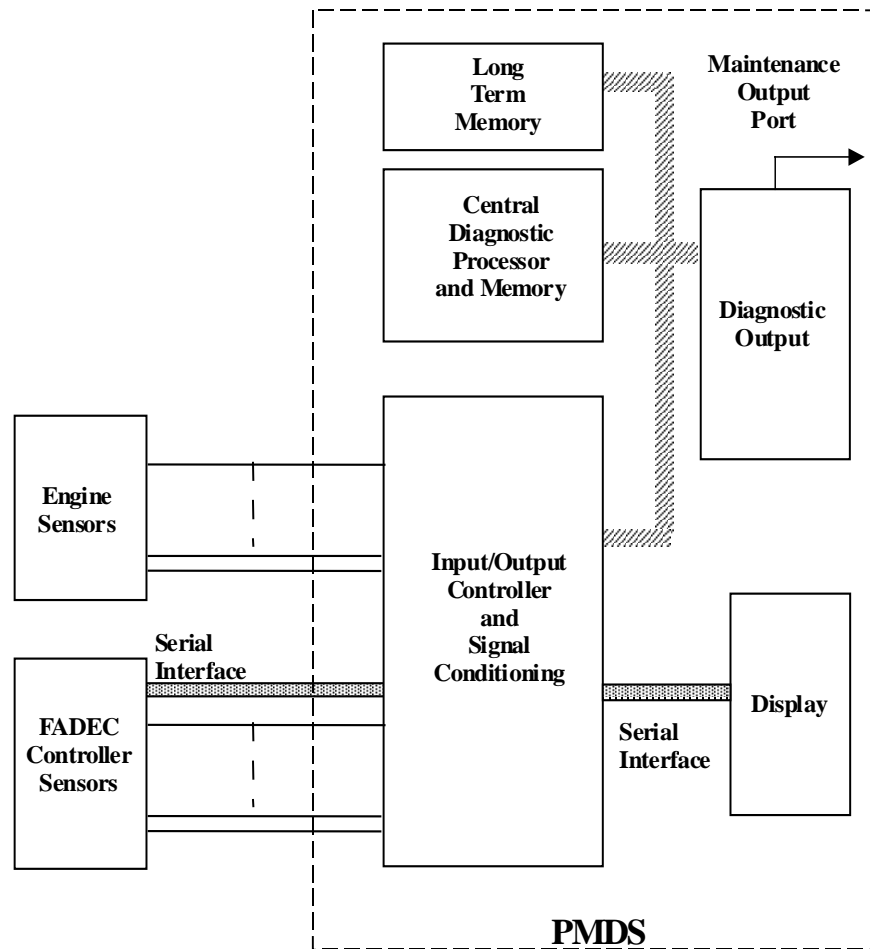


Figure 4-4. PMDS Architecture

**PMDS Hardware Design**—The PMDS hardware architecture is shown in Figure 4-5. (The hardware arrangement used in our flight testing under this program and in the earlier AGATE flight testing did not use individual engine sensors, but relied on a serial communication interface to the FADEC. The test setup is described in Section 3 of this report.) The present PMDS hardware is a prototype design implemented under the AGATE program. This PMDS hardware is shown in Figure 4-6.

During the design of the PMDS hardware in the AGATE program, Honeywell solicited guidance from the AGATE community about electronic design standards. At the time, the AGATE electronic design standards were still evolving, so the Honeywell design team opted to design the prototype PMDS hardware using then-available best practices for guidance (i.e., for lightning, EMI, thermal, shock, and other design criteria). The resulting PMDS hardware has performed flawlessly in all flight testing to date.



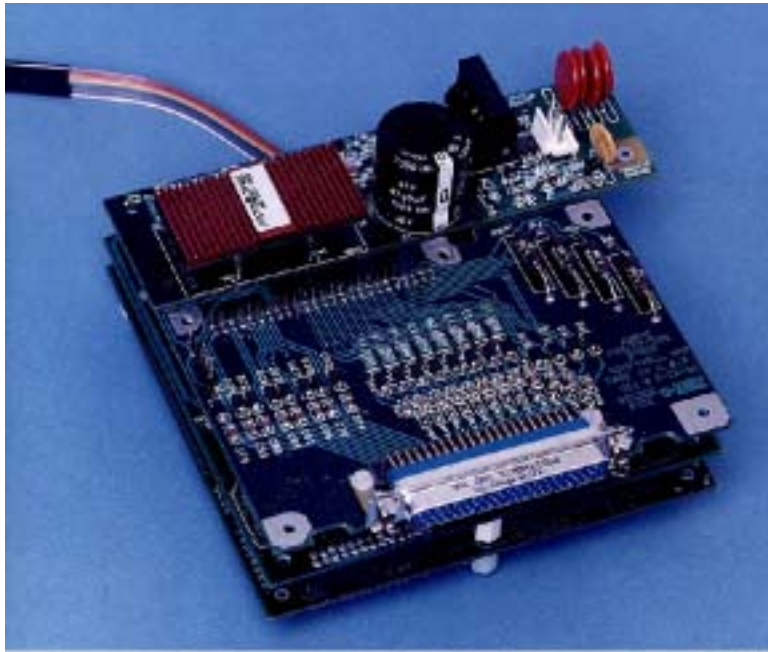
**Figure 4-5. PMDS Hardware Architecture**

**PMDS Software Design**—The present PMDS software was implemented under the AGATE program and consists of the following key elements:

- Executive program with supporting modules
- Communications to a PC for data transfer
- Communications to the FADEC for sensor data collection
- Data conversion and storage
- Built-in-test and power-up sequencing

The present PMDS software resides in two places: the run-time portion that runs on the PMDS hardware, and a retrieval portion that runs on a PC and receives saved data from the PMDS at PMDS startup.

The PMDS software runs on “bare metal,” i.e., without an operating system, as a single-thread application. At power-on, it initializes the hardware, conducts startup tests, and then sends any previously stored data out the maintenance port. A PC connected to the maintenance port and ready to receive the data will have the data sent over in a “raw” form, but be able to convert this data to a more usable form.



**Figure 4-6. PMDS Hardware**

After startup, the universal asynchronous receiver/transmitter (UART) for the port connected to the FADEC is assigned an interrupt handler and enabled, and the main thread goes into an endless loop. FADEC data is sent in records. As each byte of FADEC data arrives over the port, the interrupt handler builds the record that is being sent. When a record is complete, the main loop is given access to the record. Statistical information is computed for each field in the record and saved for later computations. With each record, a counter is incremented; when the counter reaches a threshold, the statistical information is written to the EEPROM for long-term storage.

The stored data is sent over the maintenance port whenever the PMDS powers up. To download the data, a PC must be attached via serial cable to the PMDS and the PC-side software must be running, waiting for the data to be transmitted. The data is sent in “raw” format; the PC software converts it to a human-readable format.



#### 4.2.2 Future Development

The PMDS hardware was originally designed several years ago under conditions that are markedly different than today. At that time, the hardware price goal drove designers to go with a dual-processor design using relatively low-power processors and forego expensive dual-port RAM for communication between the processors in favor of a more complicated approach with cheaper memory. These approaches and the lack of commercial alternatives led to a platform that would be difficult to upgrade in the future as processor capabilities and memory capacity increase while prices decrease.

In the time since those design decisions were made, the state of the art for small, embedded processors and cards has changed substantially. In addition to the costs of processors and memory going down and the power and capacity of processors and memory going up (Moore's Law), there are many more off-the-shelf components available. Processors substantially more powerful than the present PMDS's Motorola ColdFire 5206 are available, such as the Intel Pentium (II/III/IV) series, the PowerPC series, and MIPS and ARM. Many different single-board computers are commercially available in the PC-104 format (the same card format used in the PMDS). Also interesting is the increasing number of system-on-a-chip systems, which integrate a higher degree of functionality on a single chip, replacing functions that would otherwise be found on the system board. All of these developments can allow a more capable system for less investment. More important, they can provide the same processing capability in one processor as two ColdFire processors. Not only is this a simpler hardware design, but it also simplifies the software. One factor, however, must be kept in mind: candidate processors and boards must be chosen to meet applicable environmental requirements for temperature, humidity, and vibration.

Faster, more common processors with more memory will allow more flexibility with regard to software, both in the size of the applications and in their sophistication. The PMDS used a single-threaded application with interrupt service routines that receive data from the FADEC. This approach was dictated by the limited memory on the PMDS and the desire to keep things simple. Adding more complex algorithms to the PMDS may require a more capable infrastructure, such as an operating system (OS) might provide. Options range from open-source OSs such as eCOS (<http://sources.redhat.com/ecos/>) and Linux to commercial OSs such as Windows CE, VxWorks, and LynxOS. By using these OSs, standardized programming interfaces and off-the-shelf tools can be used.

Advancements in digital avionics will also benefit the PMDS concept. It was noted in Section 4.1 that the PMDS would benefit greatly from additional inputs outside the engine, such as aircraft speed, aileron, elevator, rudder, and flap settings, propeller pitch, etc. These other sources of data (outside the FADEC) could provide information to the PMDS over a digital avionics bus in the aircraft. This data-sharing capability will enable the use of more sophisticated models (through more sensory information) and will help to minimize the installed cost of the PMDS. The ability for different systems on board the aircraft to share digital data is a key enabler for the development and commercialization of the PMDS concept.

The future development steps described above are shown in Figure 4-7. At this time, the scope of the work, the timing, the source of development funding, and the makeup of the development team are undefined. These planning issues will be addressed as the general aviation community continues to dialog about engine diagnostics technology and market needs.

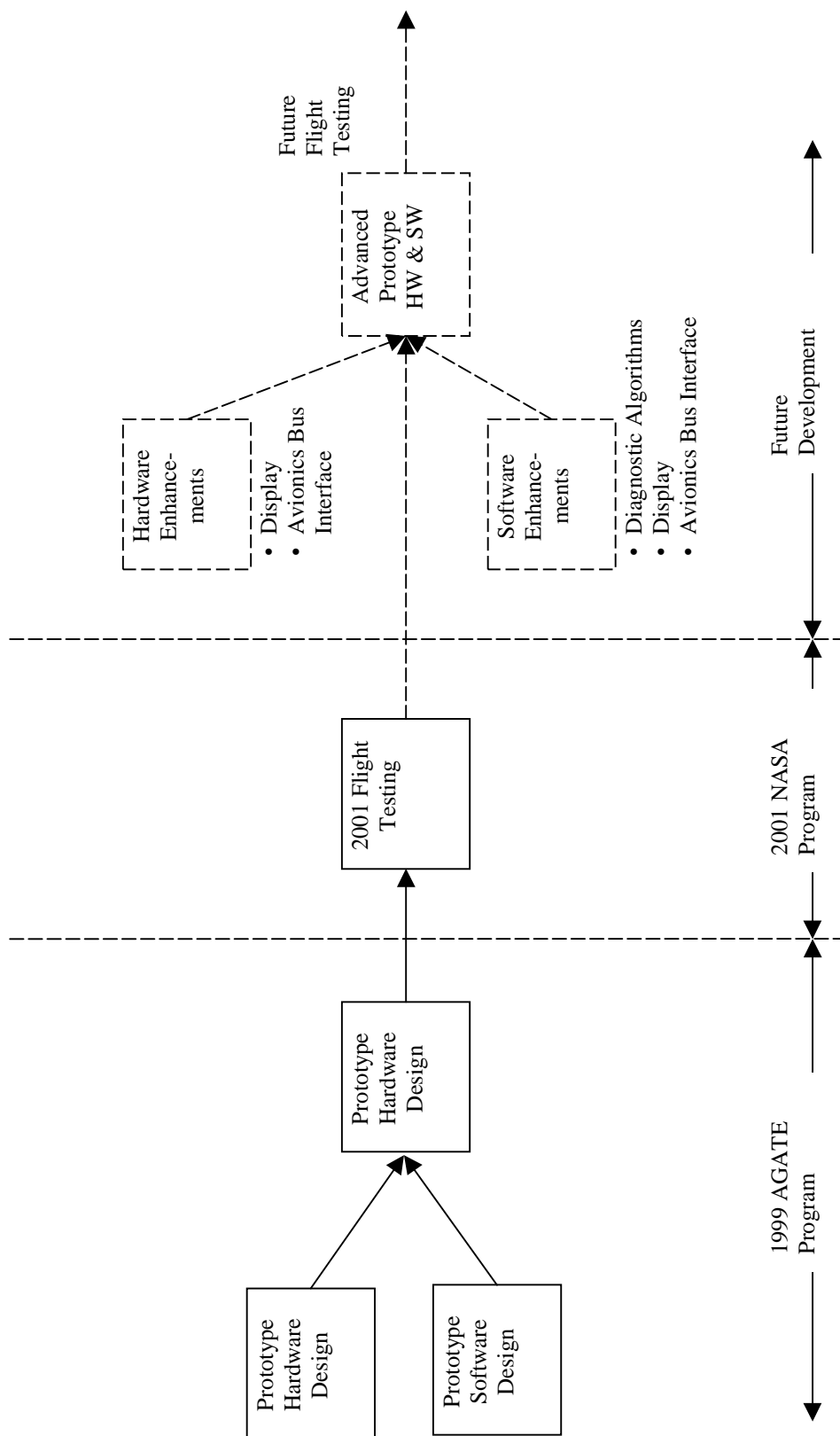


Figure 4-7. Hardware and Software Development Roadmap

## 4.3 System Development

During the AGATE program, Honeywell developed a prototype system design for the PMDS concept. This system design will require additional development to bring it to a commercializable form. The following subsections describe the key system development areas ahead.

### 4.3.1 Background

The current PMDS technology embodies the early set of system requirements defined during the AGATE program. We expect these system requirements to evolve as the development continues. Some of the key system requirements that will guide future PMDS development are

- Applications
- Certification issues
- Pilot interface
- Ground technician interface

These topics are briefly described below.

**Applications**—The PMDS flight testing performed to date has been done using the Aurora Flight Sciences Chiron aircraft with the Teledyne IO-360ES engine. Thus, all of our results are specific to that engine. Future development must, of course, address other engine models and other engine manufacturers. As market studies are made, a target market of key engine applications will be identified. These applications will be the focus of the next steps in the system development process

**Certification Issues**—The PMDS is intended strictly as a monitoring system and diagnostic-aiding system. The PMDS shall not provide test inputs to the propulsion system in order to perform. Rather, it will only monitor the propulsion system. Its outputs in flight will be limited to engine failure and warning indications. All decisions as to required pilot actions are strictly the purview of the pilot. All maintenance decisions are strictly the purview of the ground maintenance technician. The PMDS is not intended to replace these judgments. In view of these considerations, certification of the PMDS need only be to nonessential levels. The communication interface with the FADEC (and any other avionic equipment for collecting input data) will be one-way only. In this sense, the PMDS is isolated and cannot interfere with the operation of other critical systems on the aircraft. More definition of system requirements with respect to certification will be made as the development continues.

**Pilot Interface**—The PMDS shall provide two elements to the pilot interface: an engine failure indication and an engine warning indication. An engine failure indication means that the PMDS has detected an engine failure that may immediately affect the engine's power output or cause immediate harm to the engine. An engine warning indication means that the PMDS has detected an impending failure and the pilot should initiate a maintenance action before the next flight. More definition of pilot interface requirements will be made as the development continues.

**Ground Technician Interface**—During ground maintenance, the PMDS shall have a data interface for ground maintenance technicians, allowing them to interrogate the PMDS and determine fault and impending fault indications. This capability will enable technicians to download the diagnostic information that was used by the PMDS to determine the fault or warning indication. This shall include a fault history and information on the signals that caused the indications to be made. This information is

limited to fault and early detection information and is not intended to perform as a flight data recorder. The interface shall present times of failure and early detections, values of the relevant signals, details of which tests caused the indications, and an indication of the possible fault modes that led to the indication, all in a form that ground maintenance personnel can readily use.

#### **4.3.2 Future Development**

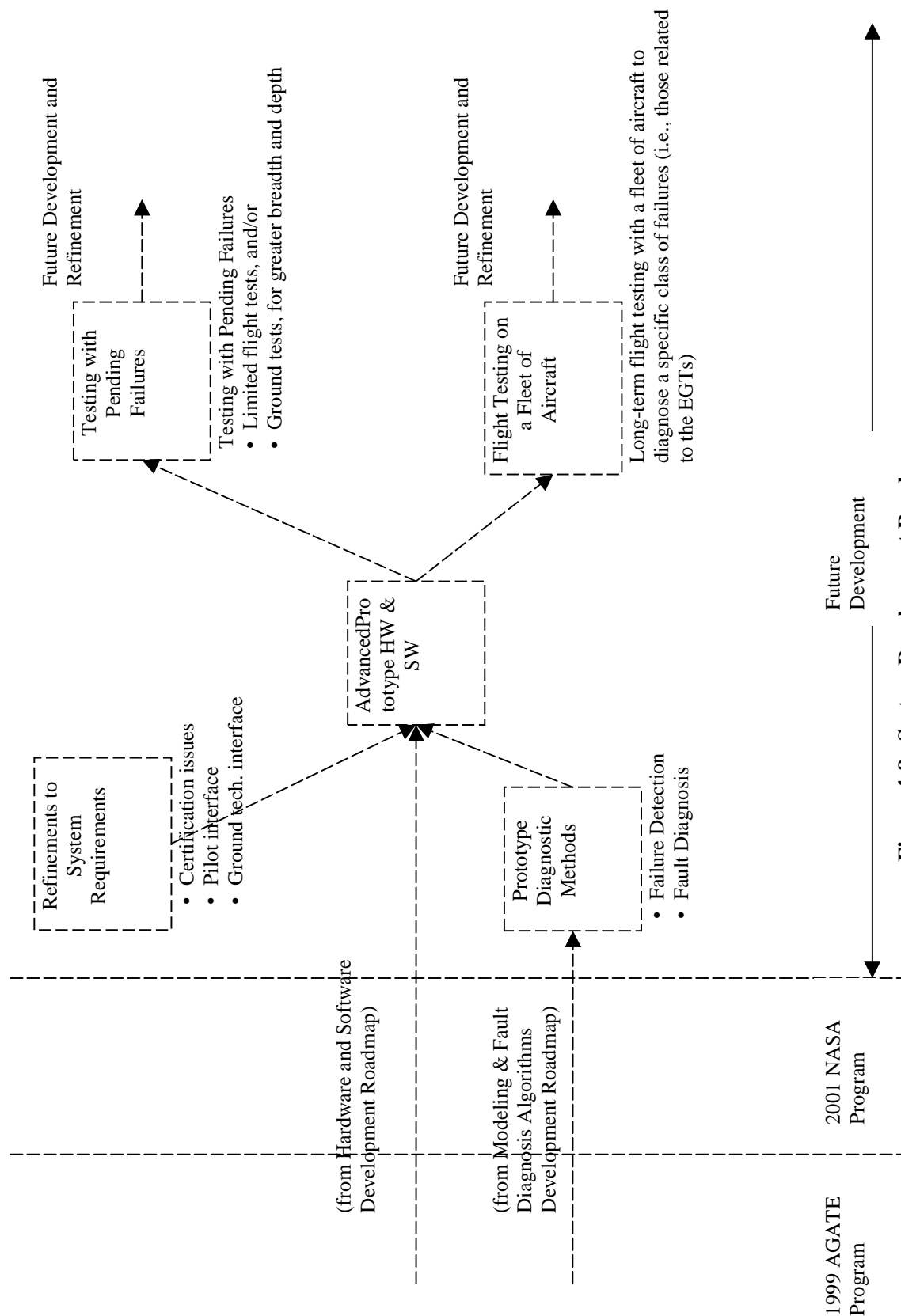
The above system requirements will be incorporated into future development work. Honeywell envisions that the next stage of development could be accomplished using two approaches: testing with a pending failure and fleet testing.

***Testing with a Pending Failure***—This type of testing would be accomplished as either:

- Limited flight tests, for failures that are not safety risks, such as failures of noncritical sensors or equipment, and/or
- Ground tests, for failures that pose a safety risk. This would also provide a means to achieve greater breadth and depth of testing.

***Fleet Testing***—This type of testing would consist of long-term flight testing with a fleet of aircraft to diagnose a specific class of failures (i.e., those that can be detected and diagnosed using data from EGTs, CHTs, and/or other recorded PMDS parameters). This type of testing will enable the development team to examine a wide range and long duration of actual operating conditions.

These two approaches could be used independently or in parallel, depending on the makeup of the development team. Future development steps using these approaches are shown in Figure 4-8. At this time, the scope of the work, the timing, the source of development funding, and the makeup of the development team are undefined. These planning issues will be addressed as the general aviation community continues to dialog about engine diagnostics technology and market needs.



**Figure 4-8. System Development Roadmap**

## **Section 5. New Technology**

Per the requirements of the NASA contract, this section is intended to "identify all nonpatentable discoveries such as improvements, innovations, and computer codes; and all patentable inventions, whether developed or discovered during performance of this contract. Possible secondary applications of reported new technology should also be included in this section."

In performing the technical work on this program, the Honeywell team did not create any new technology. As described throughout this report, our work on this program was primarily an extension of work that was begun on the AGATE program. This program's primary results are the completion of AGATE flight testing and the demonstration of the viability of PMDS technology. As such, there were no discoveries that fit any of the descriptions in the paragraph above. (During the AGATE program, Honeywell did prepare an invention disclosure covering various facets of the PMDS concept. Honeywell is currently pursuing patent protection for that intellectual property.)

## Section 6. Conclusions and Recommendations

This NASA program has taken the results of Honeywell's AGATE work and completed the initial evaluation of the capability of the prototype PMDS hardware. This work provided the following key results:

- It demonstrated the ability of the PMDS to detect a class of selected sensor hardware failures. Disconnected sensors were clearly detectable using a static correlation model that compared sensor output values with expected values. Intermittent sensor faults were also clearly detectable using a static correlation model.
- It demonstrated the ability of the PMDS hardware to successfully model the engine for the purpose of engine diagnosis. The two flight tests performed on this program demonstrated that dynamic models of the engine/aircraft can be produced using relatively simple, inexpensive instrumentation such as would be found in a commercializable on-board engine diagnostic system. Not surprisingly, nonlinear dynamic models performed better than linear dynamic models for the same number of inputs and states. Also as expected, the greater the number of test data points, the better the quality of the resulting model. (A full-scale development project would involve many sets of flight test data, thereby resulting in improved dynamic models.) Dynamic models could be used to detect faults internal to the engine as well as sensor faults.

Future development work for an engine monitoring and diagnostic system should employ the following elements:

- Engine/aircraft modeling should combine first-principles and empirical approaches. This strategy offers the advantage of using fewer parameters as required by first-principles models, while using empirical methods to calibrate unknown parameter values as needed.
- The monitoring and diagnostic system should employ additional inputs outside the engine, such as aircraft speed, aileron, elevator, rudder, and flap settings, propeller pitch, etc. This strategy will result in an improved dynamic model to be used for fault detection.
- A prioritized list of engine faults is needed to guide the diagnostic development work. Ideally, this technical information should come from an engine manufacturer. Data from more than one engine manufacturer would be even more helpful in advancing the technology.
- The monitoring and diagnostic system should be able to gather input data from the FADEC and other systems in the aircraft over a digital avionics bus. This data-sharing capability will enable the use of more sophisticated models (through more sensory information) and will help to minimize the installed cost of the PMDS.

## **Appendix**

The Appendix contains the following items:

- List of PMDS sensors
- Flight 41 documentation (consisting of Flight Test Plan 8F34, and Flight Test Engineer's Report 8F35)
- Flight 44 documentation (consisting of Flight Test Plan 8F35, and Flight Test Engineer's Report 8F36)



## PMDS Sensor Inputs

Ch.	Data	Ch.	Data	Ch.	Data
1	labels				
2	Altitude intercept	38	EGT2 intercept	74	CHT5 intercept
3	Altitude slope	39	EGT2 slope	75	CHT5 slope
4	Altitude min	40	EGT2 min	76	CHT5 min
5	Altitude max	41	EGT2 max	77	CHT5 max
6	Kollsman intercept	42	EGT3 intercept	78	CHT6 intercept
7	Kollsman slope	43	EGT3 slope	79	CHT6 slope
8	Kollsman min	44	EGT3 min	80	CHT6 min
9	Kollsman max	45	EGT3 max	81	CHT6 max
10	FuelPress intercept	46	EGT4 intercept	82	OutsideAirTemp intercept
11	FuelPress slope	47	EGT4 slope	83	OutsideAirTemp slope
12	FuelPress min	48	EGT4 min	84	OutsideAirTemp min
13	FuelPress max	49	EGT4 max	85	OutsideAirTemp max
14	OilPress intercept	50	EGT5 intercept	86	ManifoldAirPr intercept
15	OilPress slope	51	EGT5 slope	87	ManifoldAirPr slope
16	OilPress min	52	EGT5 min	88	ManifoldAirPr min
17	OilPress max	53	EGT5 max	89	ManifoldAirPr max
18	FuelTemp intercept	54	EGT6 intercept	90	AirChargeT intercept
19	FuelTemp slope	55	EGT6 slope	91	AirChargeT slope
20	FuelTemp min	56	EGT6 min	92	AirChargeT min
21	FuelTemp max	57	EGT6 max	93	AirChargeT max
22	OilTemp intercept	58	CHT1 intercept	94	EngineRPM intercept
23	OilTemp slope	59	CHT1 slope	95	EngineRPM slope
24	OilTemp min	60	CHT1 min	96	EngineRPM min
25	OilTemp max	61	CHT1 max	97	EngineRPM max
26	Bay1Temp intercept	62	CHT2 intercept	98	PressureAltitude* intercept
27	Bay1Temp slope	63	CHT2 slope	99	PressureAltitude* slope
28	Bay1Temp min	64	CHT2 min	100	PressureAltitude* min
29	Bay1Temp max	65	CHT2 max	101	PressureAltitude* max
30	Bay2Temp intercept	66	CHT3 intercept		* PressureAltitude = mbar/10
31	Bay2Temp slope	67	CHT3 slope		
32	Bay2Temp min	68	CHT3 min		
33	Bay2Temp max	69	CHT3 max		
34	EGT1 intercept	70	CHT4 intercept		
35	EGT1 slope	71	CHT4 slope		
36	EGT1 min	72	CHT4 min		
37	EGT1 max	73	CHT4 max		

## Flight Test Data Collected Manually

Ch.	Data
102	SLPC
103	Cowl flaps



## SLPC CHIRON AU-008

**N427AU****AGATE Integrated Flight Test Plan****AGATE 8F34****Operations Number:** 8F34**Range Time:** **5:00 to 7:00****Date:** **04/02/01****Proposed Engine Start:** **4:45****Estimated Flight Time:** **2.5 Hours****Flight Crew:**

Pilot in Command:

Bill Weber

Flight Test Engineer:

Ken Zugel

**Test Objectives:**

flight with PMDS, SLPC-FADEC to 4,000 ft; 8,000 ft and 12,000 ft. Several single-lever power settings. Sensors all nominal.

**Test Event Summary:****Flight - see Test Card 8F33**

Taxi.

Takeoff in SLPC mode.

Set Power = 100%.

Climb to **4,000 feet**

Set Power = 40% cruise. Reach steady state.

Set Power = 60% cruise. Reach steady state.

Set Power = 80% cruise. Reach steady state.

Set Power = 100% climb to **8,000 ft.**

Set Power = 40% cruise. Reach steady state.

Set Power = 60% cruise. Reach steady state.

Set Power = 80% cruise. Reach steady state.

Set Power = 100% climb to **12,000 ft.**

Set Power = 40% cruise. Reach steady state.

Set Power = 60% cruise. Reach steady state.

Set Power = 80% cruise. Reach steady state.

Pull power back to 40%, descend to 4,000 ft.

Set Power = 75% for enroute climb to 8,000 ft.

Level off at 8,000 ft at Power = 75%.

Continue enroute climb to 12,000 ft.

Descend to airport.

Conduct approach.

Landing.

Debrief.

**Call Signs:**

Aircraft Call Sign:

N427AU

**Frequencies:**

Manassas Tower

**Additional Information:** WX: \_\_\_\_\_**Support Equipment:** N/A**Aircraft Configuration:**

SLPC - FADEC active, FTC inactive.

MAF sensor OPERATIVE, Tbay sensor OPERATIVE  
(SLPC system). Honeywell PMDS active.

FCU Software:  
ECU Software: CH8.10  
Fuel: Full Main Tanks

**Operating Limits:** Conditions: wet/dry, daylight, VFR  
Max. altitude: 12,500 feet

**Special Precautions:** Secure Ramp Area at Aurora Hangar.



**SLPC CHIRON AU-008**

**N427AU**

**AGATE Integrated Flight Test Plan**

**AGATE 8F34**

**SIGNATURES FOR FLIGHT APPROVAL:**

Quality Assurance: \_\_\_\_\_ Date: \_\_\_\_\_.

Director of Engineering: \_\_\_\_\_ Date: \_\_\_\_\_.

Aurora FRR Board Chairman: \_\_\_\_\_ Date: \_\_\_\_\_.

Director of Flight Ops: \_\_\_\_\_ Date: \_\_\_\_\_.

Project: \_\_\_\_\_ Date: \_\_\_\_\_.

**N427AU****FLIGHT DATE: 04/12/01****Event:            Test Description:**

1. CHECKLIST PROCEDURE AND START FRONT ENGINE IN SLPC MODE.
2. START AND CHECK SLPC DATA LOG AS PER PROCEDURE.
3. START PMDS DATA LOG AS PER PROCEDURE.
4. WARMUP.
5. TAKEOFF. POWER = 100%.
6. **CLIMB TO 4,000 FEET.**
7. ADJUST COWL FLAPS AS NECESSARY, MAKE NOTES.
8. SET 40% POWER. REACH STEADY STATE.
9. SET 60% POWER. STEADY STATE.
10. SET 80% POWER. STEADY STATE.
11. SET 100% POWER, **CLIMB TO 8,000 ft.**
  
12. SET 40% POWER. REACH STEADY STATE.
13. SET 60% POWER. STEADY STATE.
14. SET 80% POWER. STEADY STATE.
15. SET 100% POWER, **CLIMB TO 12,000 ft.**
  
16. SET 40% POWER. REACH STEADY STATE.
17. SET 60% POWER. STEADY STATE.
18. SET 80% POWER. STEADY STATE.
19. SET 40% POWER OR MIN., **DESCEND TO 4,000 ft.**
  
20. SET 75% POWER, LEVEL FLIGHT.
21. INITIATE ENROUTE **CLIMB TO 8,000 ft.**
22. LEVEL OFF, CONSTANT 75% POWER.
23. INITIATE ENROUTE **CLIMB TO 12,000 ft.**
  
24. **DESCEND** TO AIRPORT.
25. APPROACH.
26. INITIATE LANDING PATTERN, LAND.
27. SWITCH OFF SLPC DATA LOG PRIOR TO FADEC SHUTDOWN.
28. SHUTDOWN.

Flight Test Engineer Report  
SLPC/PMDS Flight Test 8F35 (ref: Flight Test Plan "AGATE 8F34" 04/02/01)  
25 April 2001  
Bill Weber– Pilot  
Ken Zugel – Flight Test Engineer

Narrative:

This was a repeat of the previous flight after replacing the propeller governor. Bill and I manned the aircraft at 1215 and started the engines at 1230. We completed the checklist and powered up SLPC/PMDS system. The Manassas weather was clear, 10 miles visibility, winds were from 360 at 10 knots, and the altimeter setting was 30.28"hg. Following the runup and system checks, we took off at 1245 in SLPC mode and departed toward the southwest. We experienced problems with the landing gear system during the climbout. The gear doors did not close after the gear retracted. The gear warning horn activated during the first gear up cycle and the gear warning circuit breaker popped. Bill recycled the gear and it retracted normally and then he reset the breaker. Once the gear problems were resolved Bill continued the departure and configured the aircraft for cruise climb between 20 and 65% power. He made slow climb to 4000 ft due to airspace and air traffic control limitations. The test area was roughly between Culpeper, Charlottesville, and New Market due to air traffic concerns.

Upon reaching 4000 ft we proceeded with the cruise test points. We reversed the order of the test points from the previous flight. The altitude varied +/- 100 ft. We experienced light to moderate turbulence. The 80% power test point began at 1300 and was nominal. The cowl flaps were closed once the engine temperatures stabilized. The 60% test point began at 1306 and was nominal as well. The 40% power setting began at 1313 and was nominal as well.

The climb to 8000 ft was performed at 90% power vice the 95% on the test card due to engine/propeller limitations. The cowl flaps were opened for the climb. We leveled off at 8000 ft and started the 80% power point at 1325. The cowl flaps were closed after the temperatures stabilized. The 60% power test point was started at 1331 and was nominal. The 40% power setting was started at 1338. We did not experience any of the propeller RPM oscillations that we did on the previous flight.

The climb to 12000 ft was performed at 100% power and the climb rate varied between 300 and 500 feet per minute. Once we reached altitude and the engine temperature stabilized, the cowl flaps were closed and we started the 80% power test point at 1353. The 60% power test point was started at 1359 and was nominal. The 40% power test point was started at 1405 and was also nominal.

From the last test point Bill initiated the descent to 8000 ft at 40% power. We leveled off at 8000 ft for the 75% power test point and it was successfully completed at 1424. We continued the descent to 4000 ft at 40% power. Bill leveled off at 6000 ft to clear terrain and then continued the descent to 4000 ft once we were east of the mountains. He continued the descent at 40% power until we entered the pattern. We returned to Manassas and configured for landing. Bill made a normal landing at 1451. We taxied in and shutdown the aircraft at 1457.

Conclusions:

The SLPC system worked better than the last flight with the new propeller governor.

Recommendations:

Proceed with the next SLPC/PMDS test flight Friday at 1200.

**Proposed Engine Start:** 1200  
**Date:** 04/25/01

**Range Time:** 1230 to 1430  
**Estimated Flight Time:** 2.5 Hours

**Flight Crew:**  
 Pilot in Command: Bill Weber  
 Flight Test Engineer: Ken Zugel

**Test Objectives:** PMDS, SLPC-FADEC to 4,000 ft; 8,000 ft and 12,000 ft. Several single-lever power settings.

#### Test Card:

Power up and start data log  
 FADEC Start and Taxi.  
 Runup and test SLPC  
 Takeoff in SLPC mode.  
 Set Power = 100%  
 Reduce to 95% leaving pattern

#### Climb to **4,000 feet**

Set Power = 80% cruise. Reach steady state. Time for 5 minutes  
 Set Power = 60% cruise. Reach steady state. ""  
 Set Power = 40% cruise. Reach steady state. ""

#### Set Power = 95% climb to **8,000 ft.**

Set Power = 80% cruise. Reach steady state. ""  
 Set Power = 60% cruise. Reach steady state. ""  
 Set Power = 40% cruise. Reach steady state. ""

#### Set Power = 100% climb to **12,000 ft.**

Set Power = 80% cruise. Reach steady state. ""  
 Set Power = 60% cruise. Reach steady state. ""  
 Set Power = 40% cruise. Reach steady state. ""

Descend to 8,000 ft. at 40%,  
 Level off at 8,000 ft at 75%. Reach steady state. Time for 5 minutes.  
 Reduce power to 40%, continue descent to pattern altitude  
 Descend to airport  
 Conduct approach and Go-Around (if needed) in SLPC mode  
 Landing  
 Stop data log and shutdown  
 Debrief.

**Call Signs:** Aircraft Call Sign: N427AU  
**Frequencies:** Manassas Tower: 133.1  
 Mission at 1400: 123.45

**Aircraft Configuration:** SLPC - FADEC active, FTC inactive.  
 MAF sensor INOP, Tbay sensor INOP.  
 (SLPC system). Honeywell PMDS active.

FCU Software:  
 ECU Software: CH8.10  
 Fuel: Full Main Tanks, Full Aux. Tanks

**Operating Limits:** Conditions: wet/dry, daylight, VFR  
 Max. altitude: 12,500 feet

**Special Precautions:** Secure Ramp Area at Aurora Hangar.

SIGNATURES FOR FLIGHT APPROVAL:

Quality Assurance: \_\_\_\_\_ Date:\_\_\_\_\_.

Director of Engineering: \_\_\_\_\_ Date:\_\_\_\_\_.

Aurora FRR Board Chairman: \_\_\_\_\_ Date:\_\_\_\_\_.

Director of Flight Ops: \_\_\_\_\_ Date:\_\_\_\_\_.

Project: \_\_\_\_\_ Date:\_\_\_\_\_.

**N427AU****FLIGHT DATE: 04/25/01****Event:****Test Description:**

29. CHECKLIST PROCEDURE AND START FRONT ENGINE IN SLPC MODE.

30. START AND CHECK SLPC DATA LOG AS PER PROCEDURE.

31. START PMDS DATA LOG AS PER PROCEDURE.

32. RUNUP/ SWITCH TO FADEC MODE

33. TAKEOFF in SLPC POWER = 100%.

34. **CLIMB TO 4,000 FEET.**

35. ADJUST COWL FLAPS AS NECESSARY, MAKE NOTES.

36. SET 80% POWER. REACH STEADY STATE.

37. SET 60% POWER. STEADY STATE.

38. SET 40% POWER. STEADY STATE.

39. SET 95% POWER, **CLIMB TO 8,000 ft.**

40. SET 80% POWER. STEADY STATE.

41. SET 60% POWER. STEADY STATE.

42. SET 40% POWER. STEADY STATE.

43. SET 100% POWER, **CLIMB TO 12,000 ft.**

44. SET 80% POWER. STEADY STATE.

45. SET 60% POWER. STEADY STATE.

46. SET 40% POWER. STEADY STATE.

47. SET 40% POWER , **DESCEND TO 8,000 ft.**

48. SET 75% POWER, LEVEL FLIGHT.

49. LEVEL OFF, CONSTANT 75% POWER.

50. SET 40% POWER, **DESCEND TO 4000 ft.**

51. APPROACH AND GO-AROUND IN SLPC MODE.

52. INITIATE LANDING PATTERN, LAND.

53. SWITCH OFF SLPC DATA LOG PRIOR TO SHUTDOWN.

**54.**SHUTDOWN.



Flight Test Engineer Report  
SLPC/PMDS Flight Test 8F36 (ref: Flight Test Plan "AGATE 8F35" 04/25/01)  
27 April 2001  
Bill Weber– Pilot  
Ken Zugel – Flight Test Engineer

Narrative:

This flight was a repeat of the previous flight with the T-bay temperature and Mass Airflow sensors disconnected. The Manassas weather was 9000ft overcast, 10 miles visibility, winds were from 300 at 7 knots, the temperature was 21 degrees Celsius, and the altimeter setting was 30.08"Hg.

Bill and I manned the aircraft at 1345 and started the engines at 1401. We completed the checklist and powered up SLPC/PMDS system. Following the runup and system checks, we took off at 1416 in SLPC mode and departed toward the southwest. We didn't have any problems with the landing gear system during the climbout. Bill pulled the gear warning circuit breaker to prevent a false alarm and keep the gear down bulb from burning out as had occurred on the previous flight. Bill continued the departure and configured the aircraft for cruise climb between 75% power. He made a slow climb to 4000 ft due to airspace and air traffic control limitations. The test area was roughly between Culpeper, Charlottesville, and Fredericksburg due to ceilings.

Upon reaching 4000 ft we proceeded with the cruise test points. The altitude varied +/- 200 ft. We experienced continuous light to moderate turbulence. The cowl flaps were kept open due to the higher outside air temperatures. The 80% power test point began at 1426 and was nominal. The 60% test point began at 1432 and was nominal. The 40% power setting began at 1438 and was nominal as well.

The climb to 8000 ft was performed at 90% power due to engine/propeller limitations and resulted in a 500 fpm rate of climb. The cowl flaps were open for the climb and the entire 8000 ft test block due to the higher OAT. We leveled off at 8000 ft and started the 80% power point at 1451. The manifold pressure (MAP) stabilized at 22" Hg and the RPM was at 2450. The 60% power test point was started at 1457 and was nominal. The MAP was 21" and the RPM was 2350. The 40% power setting was started at 1503 and was nominal as well. The MAP was 20.75" and the RPM stabilized at 2325.

The climb to 12000 ft was performed at 100% power and the climb rate varied between 300 and 500 feet per minute. The indicated power command was only 96%, but appears to have been caused by a change in the Single Power Lever position sensor. We encountered wake turbulence from a B-727 descending through our altitude, which triggered several warnings and caused two momentary upsets. Once we reached altitude and the engine temperature stabilized, the cowl flaps were closed and we started the 80% power test point at 1520. The MAP was 19" and the RPM stabilized at 2525. The 60% power test point was started at 1526 and was nominal. The 40% power test point was started at 1533 and was also nominal.

From the last test point Bill initiated the descent to 8000 ft at 40% power. We leveled off at 8000 ft for the 75% power test point at 1545. Once it was completed we continued the descent at 40% power until we entered the pattern. The rate of descent was between 500 and 1000 fpm due to airspace and traffic limitations. The weather at Manassas had changed: the altimeter setting was 30.00"Hg, the skies were clear, and the temperature was 25 degrees Celsius. We returned to Manassas and made a normal landing at 1601. We taxied in and shutdown the aircraft at 1607.

Conclusions:

The SLPC system functioned nominally.

The Single Power Lever position sensor should be recalibrated if additional SLPC flights are planned.

Recommendations:

The erroneous gear warnings and malfunctions seem to be switch related and may need to be fixed prior to the next flight test.

Proceed with the next SLPC/PMDS test flight if requested and the schedule will allow it.

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